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[X] ORIGINAL [ ] REVISION NO. ______

Project No. E-16-683 (R-5936-0A0) GTRC/~~GIT~~ DATE 5 / 15 / 85

Project Director: L. W. Rehfield School/~~Lab~~ AE

Sponsor: AFOSR Bolling AFB, D.C.

Type Agreement: Grant 85-0179

Award Period: From 4/15/85 To 4/14/87 (Performance) 6/14/87 (Reports)

Sponsor Amount:

	This Change	Total to Date
Estimated:	$	$
Funded:	$ 86,000	$ 86,000

Cost Sharing Amount: $ 8,780 Cost Sharing No: E-16-345 (F-5936-0A0)

Title: Sublaminate Damage Mechanisms in Composite Structures.

ADMINISTRATIVE DATA OCA Contact Ralph Grede x4820

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Defense Priority Rating: ______ Military Security Classification: N/A
(or) Company/Industrial Proprietary: N/A

RESTRICTIONS

See Attached N/A Supplemental Information Sheet for Additional Requirements.

Travel: Foreign travel must have prior approval – Contact OCA in each case. Domestic travel requires sponsor approval where total will exceed greater of $500 or 125% of approved proposal budget category.

Equipment: Title vests with GTRC on all equipment costing less than $1,000.

COMMENTS:

Indirect Cash rate of 55.3% is fixed for the period of the Grant.

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SPONSORED PROJECT TERMINATION/CLOSEOUT SHEET

2 SR-10

Date 7-21-87

Project No. E-16-683 School/~~Lab~~ AE

Includes Subproject No.(s) N/A

Project Director(s) L. Rehfield GTRC / ~~GIT~~

Sponsor AFOSR Bolling AFB, D.C.

Title Sublaminate Damage Mechanisms in Composite Structures.

Effective Completion Date: 4/14/87 (Performance) 6/14/87 (Reports)

Grant/Contract Closeout Actions Remaining:

- [ ] None
- [X] Final Invoice or Final Fiscal Report
- [X] Closing Documents
- [X] Final Report of Inventions
- [X] Govt. Property Inventory & Related Certificate
- [ ] Classified Material Certificate
- [ ] Other

Continues Project No. Continued by Project No.

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SCHOOL OF AEROSPACE ENGINEERING

404-894-3000

DANIEL GUGGENHEIM SCHOOL OF AERONAUTICS

January 9, 1986

Dr. Anthony K. Amos
AFOSR/NA
Building 410
Bolling AFB, DC 20332

Subject: Summary of Achievements and Plans for Grant 85-0179

Dear Dr. Amos:

The subject grant is concerned with sublaminate damage mechanisms in composite structures. There are two major thrusts. The most important and the one involving the greatest risk is the creation of an approach to damage tolerance analysis and testing. What is sought here is a conceptual framework and methodology for composite structures that fracture mechanics provides for metal structures. The second thrust is the development of a mixed mode interlaminar fracture criterion for use in delamination prediction. There are activities, such as some selected experiments, which support these two major thrusts.

Considerable progress has been made on all fronts. Consequently, detailed information will be provided in slide presentation form in person in a meeting on 13 January 1986. Only the highlights will be listed and briefly discussed here.

Accomplishments

A preliminary statement of my ideas on damage tolerance analysis and testing is in Appendix I of the grant proposal. These ideas came out of our experience with delamination experiments on graphite-epoxy structures. This was a start toward thinking in a new direction. We have generalized and modified these early perceptions to the point that a new approach to damage tolerance technology for composites has been formulated and validated with static tests on a variety of structural configurations. We find that we are able to predict failure by extrapolating data from subcritical damage states. Nondestructive failure prediction is possible, then, which is a major finding.

I believe that this work shows promise of untold practical usefulness. Its novel nature prevents a precise definition of additional tasks at this time. The details will be provided in the 13 January presentation. A general direction for the second year will be indicated.

A practical working approach to analysis and design against delamination has been formulated. A simple fracture law is found to represent the existing data on four graphite-epoxy material systems and one glass-epoxy system created by a number of investigators on a variety of types of specimens. While the approach, like any engineering decision, is not unique, it has an intrinsic means of accounting for the observed high scatter and unstable character of high Mode II (or Mode III), low Mode I delamination found in experiments. Also accounted for is the unstable transition from a mixed mode fracture to an unstable Mode II shear fracture which was found in our experiments.

Interesting delamination experiments on Mode I suppression in tension, fatigue in tension and compression and tension with other complex damage modes have been conducted which support the two major thrusts.

In collaboration with Dr. Wen Chan of Bell Helicopter, a ply termination or taper experimental specimen has been conceived, designed, stress analyzed, built and tested. This is pioneering work that will provide valuable information on this common form of stress raiser and damage initiation site.

Second Year Plans

Major emphasis will be given to Damage Tolerance Technology development. Among the issues for consideration are means for predicting damage onset and final failure for damage modes with high delamination content. The base of experience and data is substantial for these forms of damage and the probability of success is high. This work will involve coordination and communication with researchers working under the auspices of AFML, AFOSR and NASA Langley Research Center.

Improvements need to be made in damage detection and tracking. Quantitative mechanical assessments of damage are needed. A promising approach is to use acoustic emission for this purpose and analyze the data using our approach to damage growth. This will be explored.

Our approach to Damage Tolerance Technology is based upon our experience with damage modes with high delamination context. Experiments and analysis will be defined and conducted on configurations with higher fiber controlled damage content. Likely candidates are plain laminates with through thickness holes or slits.

Analysis of energy release rate in the ply termination/taper specimen will be performed if time and resources permit. This is assigned secondary priority.

Closure

We are at an exciting, pivotal stage in the course of the work. Our achievements warrant continuation in our opinion. Only directions can be pinned down at present with certainty. A period of assimilation, reflection and digestion is required at this time.

Sincerely,

L. W. Rehfield
Professor and
Principal Investigator

LWR/jes

GEORGIA INSTITUTE OF TECHNOLOGY
ATLANTA, GEORGIA 30332

SCHOOL OF
AEROSPACE ENGINEERING

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OF AERONAUTICS

404-894-3000

May 9, 1986

Dr. Nicholas J. Pagano
AFOSR/NA
Building 410
Bolling AFB, D.C. 20332

Dear Dr. Pagano:

This letter follows your request for documentation of the recent contributions of my research group to the nation's technology base in advanced structures and materials. One major type of contribution is through our graduates.

At the ASTM meetings in Charleston, S.C., six former and present graduate students were on the program. The contributors were:

Ravi Deo, PhD. '77, Northrop Corporation
Ambur Reddy, PhD '80, Lockheed-Georgia Company
Pappu Murthy, PhD '81, Cleveland State University
(NASA Lewis RC)
Rao Valisetty, PhD '83, NASA Lewis RC
Erian Armanios, PhD. '84, Georgia Tech
Freddy Weinstein, PhD '86, Israel Ministry of Defense

The presentations made at the meetings appear in Table 1.

This is not an isolated instance. At the upcoming AIAA/ASME/ASCE/AHS 27th SDM Conference, four former graduate students are on the program. In addition to Drs. Armanios, Murthy and Reddy, Dr. Ronald Briley (PhD. '80, Harris Corporation) is a contributor. The presentations to be made appear in Table 2.

All of these men have been supported by AFOSR for their graduate and/or postdoctoral studies. I am very proud of them and personally satisfied that our stewardship has paid off handsomely. Please inform your management of these facts.

We continue to be grateful for AFOSR support and to work for excellent, relevant research contributions and the development of engineering talent. I am preparing a summary of our achievements for the year which will be sent separately.

Warmest regards,

L. W. Rehfield
Professor

LWR/may

cc: Dr. Michael J. Salkind
Dr. Anthony K. Amos

TABLE 1

CONTRIBUTIONS TO ASTM MEETINGS,

28 APRIL - 2 MAY, 1986, CHARLESTON, SC

MINI-SYMPOSIUM ON FRACTURE TESTING OF COMPOSITE MATERIALS STATE OF THE ART

- Deo, R.B., "Resistance Curves for Composites."

COMPOSITE MATERIALS TESTING AND DESIGN: EIGHTH SYMPOSIUM

- Murthy, P.L.N. and Chamis, C.C., "Composite Interlaminar Fracture Toughness: 3-D Finite Element Modeling for Mixed Mode I, II and III Fracture."

- Valisetty, R.R. and Chamis, C.C., "A Computer Code for Sublaminate/Ply Level Analysis of Composites and Interlaminar Fracture Toughness of End-Notch and Mixed-Mode Fracture Specimens."

- Armanios, E.A. and Rehfield, L.W., "Interlaminar Fracture Analysis of Composite Laminates Under Bending and Combined Bending and Extension."

- Reddy, A.D., Rehfield, L.W., Weinstein, F. and Armanios, E.A., "Interlaminar Fracture Processes in Resin Matrix Composites under Static and Fatigue Loading."

TABLE 2

CONTRIBUTIONS TO 27TH AIAA/ASME/ASCE/AHS

STRUCTURES, STRUCTURAL DYNAMICS AND MATERIALS CONFERENCE

19-21 MAY, 1986, SAN ANTONIO, TX

- Laurenson, R.M., Hauenstein, A.J., Gubser, J.L. and Briley, R.P., "Effects of Structural Nonlinearities on Limit Cycle Response of Aerodynamic Surfaces," AIAA-86-0899-CP.
- Rehfield, L.W. and Reddy, A.D., "Observations on Compressive Local Buckling, Postbuckling and Crippling of Graphite/Epoxy Airframe Structure," AIAA-86-0923-CP.
- Chamis, C.C., Aiello, R.A. and Murthy, P.L.N., "Fiber Composite Sandwich Thermostructural Behavior: Computational Simulation," AIAA-86-0948-CP.
- Rehfield, L.W. and Armanios, E.A., "Interlaminar Fracture Analysis of Laminated Composites Using a Sublaminate Approach," AIAA-86-0969-CP.

E-16-683

GEORGIA INSTITUTE OF TECHNOLOGY
ATLANTA, GEORGIA 30332

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OF AERONAUTICS

404-894-3000

June 10, 1986

Dr. Anthony K. Amos
AFOSR/NA
Building 410
Bolling AFB, D. C. 20332

Subject: Annual Scientific Report for Grant No. AFOSR-85-0179
(Georgia Tech Project E16-683)

Dear Dr. Amos:

A complete briefing was given to Dr. Nicholas Pagano on 13 January 1986 on the research performed under the subject grant. At that time, a summary of achievements and plans was presented in the attached letter dated 9 January. In response to a request by Dr. Pagano, the attached letter dated 9 May was prepared which summarizes recent contributions by my former and current students.

In response to another request by Dr. Pagano, the attached "achievement" package was prepared. This highlights a specific finding of our research.

Our delamination experiments on Mode I suppression in tension, fatigue in tension and fatigue in compression are summarized in the enclosed manuscript. The Mode I suppression work provides a key to delamination management and control in structures. Our modeling success here is an achievement of which we are most proud.

Currently, we are searching for the physical basis for the increasing resistance to cracking in tension that has been found in our experiments. The mechanics models that we have tried to date are inadequate. An understanding of this issue is very important and, with the concurrence of Dr. Pagano, we are giving it high priority.

Sincerely,

L. W. Rehfield
Professor

LWR/may
Attachments

Presented at the ASTM Eighth Symposium
on Composite Materials Testing and Design
Charleston, South Carolina
28-30 April 1986

INTERLAMINAR FRACTURE PROCESSES IN RESIN MATRIX COMPOSITES UNDER STATIC AND FATIGUE LOADING*

A. D. Reddy**, L. W. Rehfield, F. Weinstein and E. A. Armanios†
School of Aerospace Engineering
Georgia Institute of Technology
Atlanta, Georgia 30332

ABSTRACT

Both static and fatigue test results on a double-cracked-lap-shear specimen are presented under tensile and compressive loading. The specimen is made of AS4/3501-6 graphite/epoxy material with a $(0,90,\pm45)_s$ quasi-isotropic balanced symmetric layup in the lap and strap. In order to suppress the mode I contribution in this specimen which exhibits mixed mode behavior in tensile loading, a normal force is applied close to the delamination front. The response due to this force is estimated from a sublaminate theory which accounts for transverse shear strain and normal and axial displacements. The normal force application is effective only when the delamination growth is dictated by mixed mode behavior. On the basis of the limited experimental data generated thus far, it seems that the static and fatigue delamination growth thresholds are indistinguishable, though cyclic tensile loading induces larger delamination growth. Considerable improvement in fracture behavior was exhibited by mode I suppression under both tensile and fatigue loading. This partially explains the effectiveness of stitching and wrapping free edges in practical structures.

KEY WORDS: composite materials, composite structures, graphite-epoxy, crack propagation, fracture, fatigue, delamination, test methods.

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* Work sponsored by AFOSR under Grants 83-0056 and 85-0179.
** Presently Specialist Engineer at Lockheed-Georgia Company, Marietta, Georgia.

INTRODUCTION

Interlaminar fracture or delamination is a primary damage mode in laminated composites. It is caused by high interlaminar stresses which are produced by local stress raisers which may be manufacturing-related or service induced. Delaminations alter internal load paths and usually contribute to the ultimate failure of the structure. Interlaminar fracture toughness, which characterizes the resistance to delamination, is a key parameter in describing the damage tolerance of laminated composite materials. A material characteristic that is a measure of fracture toughness and that is often used to determine resistance to delamination growth is the strain energy release rate G. It can be obtained for three particular modes of crack action - - - as mode I or the opening mode, mode II or the forward shearing mode and mode III, the tearing mode.

Investigation of the strain energy release rate associated with mode I fracture using the double cantilever beam and an edge delamination specimen have been reported by several investigators [1-7]. Mixed-mode interlaminar tests include the single cracked-lap shear specimen [7-8], the edge delamination specimen [9], the mixed mode flexural specimen [7], and the double cracked lap shear specimen [10]. A popular test to determine mode II interlaminar fracture toughness is based on the use of the end-notched flexural specimen [7].

The double cracked-lap-shear specimen [10] has been designed such that both tension and compression tests can be performed on the same specimen. Tests on graphite-epoxy specimens have shown that a stable crack growth results under tensile loading followed by sudden failure, whereas a single, unstable, catastrophic fracture event occurs under compression. This specimen fails in a mixed mode under tensile loading and mode II under compressive loading. It has been analyzed using simple analytical theories [10]-[11] which can be used for parametric studies of interlaminar fracture processes. Therefore, a similar double cracked-lap shear specimen is used in the present research. Both static and fatigue loading situations under both tension and compression have been studied.

A weakness of laminated composite construction is that there is no transverse or through thickness reinforcement. The delamination can easily propagate along the interface between two adjacent layers. In graphite-epoxy systems under mixed mode situations, it was found that G_{II_c} is higher than G_{I_c} [7]. The mode I or the peel stress-induced failure mode, is the one that drives the delamination [12]. On the basis of limited tests, it has been noted that mode I is also the one that drives the disbond in bonded lap joints [12][14]. One might expect that by suppressing mode I a considerable improvement in interlaminar strength can be achieved. Consequently, the first part of this experimental and analytical study focuses on opening mode effects and their suppression.

The second part of the study deals with cyclic delamination growth under constant amplitude tension-tension and compression-compression fatigue loading. This is the only known compression-compression fatigue interlaminar fracture work of its kind. Data are generated first on the double crack lap shear (DCLS) specimens fabricated out of both AS4/3501-6 and AS4/3502 graphite-epoxy material systems. The mode I suppression methodology is used next in the fatigue loading context to study the cyclic delamination growth retardation. As static compression loading on the specimen causes failure only in mode II, the compression fatigue tests are performed mainly to confirm the sudden, total failure of the specimen and also to determine the threshold levels.

TEST SPECIMENS

A single panel was fabricated by the Lockheed-Georgia company and inspected for quality using standard aerospace industry practice before sectioning into twenty test specimens. During layup, folded Kapton film was placed at the end of the lap/strap interface to initiate a delamination under subsequent loading. Figure 1 shows the dimensions of the tested specimens. The nominal mechanical properties of AS4/3501-6 and AS4/3502 graphite-epoxy tape are listed Table 1.

EXPERIMENTAL PROCEDURE

The static tension tests on the DCLS specimens were performed on a displacement controlled Baldwin screw-type testing machine. A sheet of photoelastic material was bonded on each of the surfaces of the specimen lap portion. Isochromatic fringes develop at the crack front as a result of the high strain gradient in that vicinity when the specimen is loaded. These fringes were tracked through the analyzer of a reflection polariscope to locate the crack front. Also, the isochromatic fringes served to check initial uniformity of the applied load on the specimen. The specimen was also instrumented with a custom-built, linear variable differential transformer (LVDT) displacement transducer-based extensometer to measure its compliance.

The specimen was loaded continuously and the cracks on both faces were followed through separate polariscopes. At a certain critical load, the first threshold of the crack growth occurred at the lap/strap juncture and it grew beyond the Kapton film. With crack growth, the specimen stiffness reduces and hence the load drops in a controlled displacement machine. Loading was stopped whenever crack growth was observed and the extent of propagation was recorded together with the load. This procedure is described in Ref. [10]. The specimens were carefully positioned and the tab ends were tightly held between serrated grips. The load was applied at a displacement control rate of 0.254 mm per minute.

Mode I suppression is achieved by applying a clamping force normal to the delamination front. A calibrated C-clamp type device (Figure 2) is used to apply the normal loading (which tends to produce compressive peel stress) on the specimen. The movement of the crack front is monitored through the photoelastic coating bonded on the lap surface of the specimen. The clamp is positioned slightly ahead of the crack front and tightened to apply a known initial normal loading. The specimen is then loaded in tension. The load at which the crack crosses the normal loading line is recorded together with the crack growth. This procedure is repeated until the laps delaminate with sudden, unstable crack growth.

The fatigue tests on the DCLS specimens were conducted on a displacement controlled Instron screw-type testing machine at a frequency of 23 cycles per minute. Constant-amplitude loading with a stress ratio of 0.2 was applied, where the stress ratio is defined as the ratio of the minimum to maximum stress in the load cycle. The same mode I suppression methodology as in static loading was used in the fatigue constant amplitude tension-tension situations. The number of cycles at which the delamination extended was also recorded additionally.

Altogether, four DCLS specimens were tested to investigate the influence of mode I suppression on the static behavior, and four specimens were tested to investigate the fatigue loading behavior.

ANALYTICAL APPROACH

The static analysis of the DCLS specimen is based on the shear deformation approach developed in Ref. [11]. This approach will be used in order to conduct a mode I suppression investigation on the DCLS specimen.

The shear deformation (SD) model includes transverse shear strain, axial and normal displacement distributions. For laminated composite structures, this approach can be applied by a ply-by-ply basis or to a group of plies (sublaminates) as if they are homogeneous bodies in equilibrium. Interfacial stresses are initially unknowns. Enforcement of continuity conditions at the interface and boundary conditions in an overall sense leads to a solution for these stresses. A complete development of the analytical model is provided in Ref. [11]. Due to the symmetry, only one half of the structure is analyzed. In this formulation, the specimen will be considered as composed of four sublaminate elements as shown in Figure 3, where the global coordinates are x and z. Each element is treated as a homogeneous orthotropic sublaminate characterized by its laminate properties.

The SD model is used to estimate the relation between the applied load P and the normal load F needed to suppress the opening mode. The condition corresponding to total mode I suppression is that the vertical displacement W below the normal load is zero. An important factor that influences the ratio P/F is the distance "a" along the cracked overhang portion on which F is applied.

Using the SD model, the axial forces, moments and displacement distributions are obtained for each sublaminate. A summary of the governing equations and selected relations are given in the Appendix for completeness. Finally, the vertical displacement of sublaminate 4 is obtained as a function of the applied loads P, F, the distance "a" and the geometry and elastic properties of sublaminates 1, 2, 3 and 4. By assuming that the elastic properties of sublaminates 1 and 4 are the same, the vertical deflection at $x = \ell$ is

$$W = a\left[\frac{F}{A_{55}} + \frac{Fa^2}{3D_{11}} - \frac{N_{1j}M_{1j}\sinh s_j(\ell-a)}{s_jD_{11}}\right](j = 1,2) \qquad (1)$$

Summation over the index j is implied in the third term of Equation (1). The constants represented by the s_j, N_{1j} and M_{1j} are defined in the Appendix.

By setting the vertical displacement given by Equation (1) equal to zero, a trivial answer is obtained for a = 0. For the case a ≠ 0, the ratio P/F is

$$P/F = \frac{\frac{D_{11}}{A_{55}} + \frac{a^2}{3} + \frac{a}{M_{11} - M_{12}}\frac{M_{1j}}{s_j}\tanh s_j(\ell-a)}{\frac{A_{11}}{2A_{12}}\;\frac{M_{11}M_{12}}{M_{11} - M_{12}}\;\frac{\tanh s_j(\ell-a)}{s_j}} \qquad (2)$$

where A_{12} is defined in the Appendix.

RESULTS AND DISCUSSION

Mode I Suppression

Figure 4 shows the relation between the ratio P/F and the distance "a", given by Equation (2). This equation does not predict the ratio P/F when a = 0 for which the trivial answer is achieved. The assumption that mode I is suppressed by equating the vertical deflection below the normal load F to zero is appropriate for a small crack length. The results are obtained using the properties for the AS4/3501-6 material system given in Table I.

For the DCLS specimen an assumed value for "a" of 0.254 to 0.762 mm (0.01 to 0.03 inches), which represents 4-12 percent of the crack length, is adequate. In this range, the values for the ratio P/F obtained using Figure 4 are 80 to 100. Using Equation (2), a linear relation between P and F is obtained for a certain value of "a".

The mode I suppression study centered upon experiments conducted by applying lateral loads, in addition to the direct tensile loads, to close the delamination cracks and arrest crack growth. The test procedure for each successive specimen has been modified to account for findings in the previous tests. In particular, the lateral load levels are selected in this way.

Figure 5 summarizes the results obtained from mode I suppression tests. The static critical load at which the delamination starts to propagate is denoted by P_{cr}. Its value was obtained by averaging data from five specimens, three of AS4/3502 and two of AS4/3501-6 statically tested. The failure load ranged from 48.8-53.4 KN (10960-12000 lb) with a mean value of 50.7 KN (11400 lb). There is no significant difference between the tests of the two materials as presented in Table 2. The resistance curve from the data obtained under tensile loading is represented by the full line AB. In the first mode I suppression test a normal load of 396 N (89 lb) increased the value of P_{cr} by about 27 percent to 63.1 KN (14200 lb). At this load the delamination propagated to 3.81 mm (0.15 inches) followed by a small load drop. When the normal load was increased to 444N (100 lb) an additional delamination of 3.81 mm (0.15 inches) was observed at 65.6 KN (14750 lb). This procedure was repeated for different values of F as shown in this figure. After a delamination growth of about 7.62 mm (0.3 inches) the normal load F does not affect the load-crack growth dependence as shown by the resistance line.

In Ref. [16] three phases in the resistance curve for DCLS specimens under tensile load were defined. A starting region corresponding to the onset of the delamination growth , an intermediate region corresponding to stable delamination growth and a final or failure region of unstable delamination growth. It is observed that the loads corresponding to the unstable delamination growth under tensile and compressive loading are nearly the same. This specimen fails under an unstable crack growth (mode II behavior) under compressive loading. The similarity between the fracture surfaces in the unstable regions under both loading conditions suggest that the unstable phases are mode II controlled. Moreover, the mixed-mode ratio in the specimen may vary along the resistance curve starting at the initial ration (G_I/G_{II} = 66%) and ending with a mode II controlled behavior.

It appears from Figure 5 that for specimen 1 the effect of increasing the applied normal force F above 444 N (100 lb) does not result in much improvement in the fracture toughness beyond the linear trend line. From the previous tension test on DCLS specimens [16], it was found that at an average total delamination length of 7.62 mm (0.3 inches) the behavior transitioned from stable mixed mode to unstable mode II behavior. A similar change in modes in the present tests would have resulted in this type of behavior as the normal load cannot affect the mode II contribution.

On a second specimen, a normal load of 680.5 N (153 lb) was applied and an increase of P_{cr} to 71.2 KN (16000 lb) was observed. The measured delamination growth was 2.54 mm (0.1 inches) which is below the stable region limit of 7.62 mm (0.3 inches). An increase in the normal load to 845 N (190 lb) totally arrested the delamination growth until the final failure load was reached.

The normal load was increased to 845 N (190 lb) on a third specimen and P_{cr} ascended to 74.2 KN (16680 lb). At this stage a delamination growth of 1.01 mm (0.04 inches) was recorded. On this specimen a normal load of 890 N (200 lb) suppressed the delamination growth until the final failure. The total contribution of mode I was eliminated on the fourth specimen when a normal load of 890 N (200 lb) was applied. In this case, the applied tensile loading reached the final failure load and the unstable delamination growth occurred. This failure behavior is similar to the compression loading when no delamination growth was observed under increasing load until the ultimate load was reached.

This mode I suppression study provides a partial explanation for the effectiveness of stitching in laminates and stiffener/skin construction when the opening mode is suppressed or retarded. The benefits are due to the fact that $G_{II_c} >> G_{I_c}$ for the graphite-epoxy material systems.

Figure 6 shows the dependence of F on the applied load P_{cr} needed to initiate the delamination growth. A linear relationship between P and F is observed. This is in agreement with the linear behavior obtained by using Equation 2. The total mode I suppression is achieved at a critical load of 78.3 KN (17600 lb) with a normal load of 890 N (200 lb), which represents a P/F ratio of 88. This ratio agrees with the predicted value of 80 to 100 obtained for small values of "a" using Equation (2).

The experimental evidence and the analytical predictions demonstrate the effectiveness of mode I suppression for graphite-epoxy material systems. Diminishing benefits are found as pure mode II behavior is approached, as would be logically expected.

Constant Amplitude Fatigue

The constant amplitude tension-tension fatigue test are presented in Figure 7. The term P_{max} denotes the maximum applied cyclic load. At R = 0.2 and P_{max}/P_{cr} = 0.85, no delamination was observed after 200 cycles. In an effort to grow the delamination, the ratio P_{max}/P_{cr} was increased to 0.95. The value of P_{max} lies within the scatter band of P_{cr} from the static tensile tests. Under this loading, delamination grew steadily and reached an asymptotic value of 8.63 mm (0.34 in.) at 1500 cycles. Though the cyclic tensile load threshold is indistinguishable from the static value, it results in larger delamination growth. More data is needed in order to establish a reliable conclusion regarding the fatigue delamination threshold.

A second specimen was tested at the same P_{max}/P_{cr} ratio, but with total mode I suppression, applying a normal load of 890 N (200 lb). This test was terminated at 19000 cycles as no delamination growth was observed. This demonstrated the tremendous improvement in delamination growth retardation in fatigue loading by suppressing mode I totally.

On a third specimen, a partial mode I suppression was imposed by applying a normal load of 680N (153 lb). It needed 6200 cycles to grow the delamination to a length of 8.63 mm (0.34 in.) compared to 1500 cycles in the specimen with no mode I suppression. The test was continued on this specimen after removing the mode I suppression setup and an asymptotic value of 15.24 mm (0.6 in.) was reached at 20,000 cycles.

The fatigue compression-compression test results are presented in Figure 8. The load at which the failure occurs under static compression loading is denoted by P_{ULT}. No delamination growth was observed at a ratio $P_{max}/P_{ULT} = 0.8$ after 5200 cycles. In an effort to grow the delamination this ratio was increased to 0.9, and a catastrophic failure similar to the static behavior occurred at 14,700 cycles.

The fatigue experiments suggest that static and fatigue behavior are similar. Of particular practical importance is the fact that stable crack growth under compression does not occur in these materials either static or fatigue loading. Also, for a given maximum tensile load level, the delamination is larger in fatigue than under static conditions.

CONCLUDING REMARKS

Interlaminar fracture processes in AS4-3501-6 and AS4/3502 graphite-epoxy material systems have been investigated using DCLS specimens. This paper summarizes the experimental information and analytical results on mode I suppression and the experimental data under both tension-tension and compression-compression fatigue loading.

The "post delamination" behavior plays an important role in the effectiveness of the mode I suppression procedure. Mode I suppression can improve fracture toughness in delamination growth regions dominated by mixed mode behavior. Crack growth close to an unstable region is controlled by mode II behavior, which cannot be arrested by mode I suppression. The experimental results confirm that mode I has a major influence in fracture processes in laminated composite under mixed mode situations for graphite-epoxy material systems.

The fatigue tests suggest that cyclic delamination growth can be retarded or totally suppressed by using mode I suppression methodology. Though the cyclic tensile load threshold is indistinguishable from the static value, it results in larger delamination growth or, higher growth rates.

For the DCLS specimen a linear relation was obtained between the normal load applied to suppress mode I and P_{cr}. The SD model used to evaluate the P/F ratio needed to suppress mode I contribution gives satisfactory results and is in accordance with the test data within 9 to 14 percent. This model is simple and very suitable for performing parametric design-related studies where a large number of configurations are to be evaluated quickly and economically.

Table 1. Properties of Graphite/Epoxy Tape AS4/3501-6 and AS4/3502

Material Tape	E_{11} GPa(MSI)	E_{22} GPa(MSI)	G_{12} GPa(MSI)	ν_{12}	Nominal Fiber Volume %	Nominal Ply Thickness mm(inches)
AS4/3501-6	139.3(20.2)	11.7(1.70)	5.8 (0.85)	0.3	65%	0.1397(0.0055)
AS4/3502	141.3(20.5)	11.5(1.67)	6.0 (0.87)	0.28	65%	0.1397(0.0055)

Table 2. Summary of Experimental Static Data on DCLS Specimens

Specimen		P_c		P_f		G_c		G_f	
Number	Material	N	(lb)	N	(lb)	J/m^2	$(in\cdot lb/in^2)$	J/m^2	$(in\cdot lb/in^2)$
1(T)	AS4-3502	53379	(12000)	79174	(17800)	396.7	(2.265)	872.0	(4.979)
2(T)	AS4-3502	48753	(10960)	68500	(15400)	343.8	(1.963)	679.0	(3.877)
3(T)	AS4-3502	49731	(11180)	66630	(14980)	360.6	(2.059)	647.5	(3.697)
4(C)	AS4-3502	64496	(14500)	64496	(14500)	581.8	(3.322)	581.8	(3.322)
5(C)	AS4-3502	77662	(17460)	77662	(17460)	857.3	(4.895)	857.3	(4.895)
6(C)	AS4-3502	80064	(18000)	80064	(18000)	996.0	(5.687)	996.0	(5.687)
7(T)	AS4-3501-6	51596	(11600)	85400	(19200)	381.1	(2.176)	1044.2	(5.962)
8(T)	AS4-3501-6	50040	(11250)	76330	(17160)	359.0	(2.050)	834.0	(4.762)
9(C)	AS4-3501-6	67600	(15200)	67600	(15200)	654.3	(3.736)	654.3	(3.736)

(T) and (C) denote tensile and compression test, "c" and "f" denote starting critical value and final failure, respectively. Data for specimens 1-6 are taken from Ref. [16].

APPENDIX

Shear Deformation Model

Consider a laminate made of N perfectly bonded layers or plies, each ply having a plane of material symmetry parallel to the plane of the laminate in a state of plane strain. Select a sublaminate or a particular ply that is singled out for study as shown in Figure 9. The thickness is h. The interlaminar stresses τ_{zz} and τ_{xz} at the top surface are denoted by P_2 and T_2, respectively, while the corresponding stresses at the bottom of the ply are designated as P_1 and T_1.

The basic equations for the shear deformation model encompass the following governing equations for each sublaminate or group of plies:

Overall Equilibrium

$$\begin{aligned} &N_{,x} + n = 0 \\ &Q_{,x} + q = 0 \\ &M_{,x} - Q + m = 0 \end{aligned} \tag{3}$$

where n, q, m can be regarded as an effective distributed axial force, lateral load and moment, respectively. They are defined as

$$\begin{aligned} &n = T_2 - T_1 \\ &q = P_2 - P_i \\ &m = \frac{h}{2}(T_1 + T_2) \end{aligned} \tag{4}$$

N, Q and M are the axial stress, shear stress and moment resultants, respectively.

Constitutive Relationships

$$
\begin{aligned}
N &= A_{11}U_{,x} + B_{11}\beta_{,x} \\
Q &= A_{55}(\beta + W_{,x}) \\
M &= B_{11}U_{,x} + D_{11}\beta_{,x}
\end{aligned}
\tag{5}
$$

where

$$
\begin{aligned}
(A_{11}, B_{11}, D_{11}) &= \int_{-h/2}^{h/2} C_{11}(1, z, z^2)dz \\
A_{55} &= \int_{-h/2}^{h/2} C_{55}\, dz
\end{aligned}
\tag{6}
$$

and the stiffness matrix C_{ij} represents the relations between the stress and strain fields given by Equation (7) for plane strain situation.

$$
\begin{Bmatrix} \tau_{xx} \\ \tau_{zz} \\ \tau_{xz} \end{Bmatrix} = \begin{bmatrix} C_{11} & C_{13} & 0 \\ C_{13} & C_{22} & 0 \\ 0 & 0 & C_{55} \end{bmatrix} \begin{Bmatrix} \varepsilon_{xx} \\ \varepsilon_{zz} \\ \gamma_{xz} \end{Bmatrix}
\tag{7}
$$

Displacement Distributions

The axial displacement u and the transverse displacement w are assumed to be of the form

$$u(x, z) = U(x) + z\beta(x)$$

$$w(x) = W(x) \qquad (8)$$

where $U(x)$ is the axial displacement along $z = 0$, β is the shear angle and $W(x)$ is the displacement in z direction.

Based upon these assumptions an analytical solution can be obtained for the DCLS specimen. The specimen is divided into four sublaminates as shown in Figure 3, and the interlaminar boundary conditions of tractions and displacements continuity are applied at the interface between sublaminates 1 and 2. These boundary conditions, in addition to the symmetry conditions along x axis, enable one to express the variables associated with sublaminates 1 and 2 in terms of the axial stresses and moment resultants N_1, N_2, M_1, M_2. The net result yields four coupled differential equations as developed in Ref. [11]. Their solution leads to the following characteristic equation.

$$s[B_1 s^4 - B_2 s^2 + B_3] = 0 \qquad (9)$$

where s is the characteristic root and

$$B_1 = B_3 \left(\frac{D_{11}}{A_{55}}\right)_1 \left(\frac{D_{11}}{A_{55}}\right)_2 + \left(\frac{D_{11}}{A_{55}}\right)_2 \left(\frac{h^2}{A_{55}}\right)_1 + \left(\frac{D_{11}}{A_{55}}\right)_1 \left(\frac{t^2}{A_{55}}\right)_2$$

$$B_2 = B_3 \left[\left(\frac{D_{11}}{A_{55}}\right)_1 + \left(\frac{D_{11}}{A_{55}}\right)_2\right] + \left(\frac{h^2}{A_{55}}\right)_1 + \left(\frac{t^2}{A_{55}}\right)_2 \qquad (10)$$

$$B_3 = 4 \left[\left(\frac{1}{A_{11}}\right)_1 + \left(\frac{1}{A_{11}}\right)_2\right]$$

The boundary conditions associated with the sublaminate sections are given below for constant values of x.

At $x = 0$: $u_1 = u_2 = 0$

$\beta_1 = \beta_2 = 0$

At $x = \ell - a$: $u_2 = u_3$

$\beta_2 = \beta_3$

$N_1 = M_1 = 0$ (11)

$N_2 = N_3$

$M_2 = M_3$

At $x = d$: $N_3 = \frac{P}{2}$

$M_3 = 0$

At $x = \ell$: $Q_4 = -F$

$M_4 = 0$

Subscripts 1-4 denote the sublaminate number. The constants resulting from the solution of the governing equations are obtained using Equation (11).

The axial forces related to sublaminate 1, 2 and 3 are:

$$N_1 = N_{1j} \cosh s_j x + N_o \tag{12}$$

$$N_2 = - N_{1j} \cosh s_j x + \frac{(A_{11})_2}{(A_{11})_1} N_o \tag{13}$$

$$N_3 = \frac{P}{2} \tag{14}$$

where

$$N_o = \frac{P}{2} \frac{(A_{11})_1}{A_{12}} \qquad (15)$$

$$N_{11} = \frac{N_o M_{12} - Fa}{(M_{11} - M_{12})\cosh s_1(\ell - a)} \qquad (16)$$

$$N_{12} = \frac{N_o M_{12} + Fa}{(M_{12} - M_{11})\cosh s_2(\ell - a)} \qquad (17)$$

$$M_{1j} = \frac{\frac{hs_j^2}{2}\left(\frac{D_{11}}{A_{55}}\right)_1}{\left[1 - s_j^2\left(\frac{D_{11}}{A_{55}}\right)_1\right]} \qquad (18)$$

$$A_{12} = (A_{11})_1 + (A_{11})_2 \qquad (19)$$

Summation over the index j is implied in Equations (12), (13) and (18). Finally, the shear and peel stresses at the interface between sublaminates 1 and 2 are obtained as:

$$T_1 = s_j N_{1j} \sinh s_j x \qquad (20)$$

$$P_1 = s_j^2 (M_{1j} + \frac{h}{2}) N_{1j} \cosh s_j x \qquad (21)$$

ACKNOWLEDGEMENTS

This work was supported by the U.S. Air Force Office of Scientific Research under Grants 83-0056 and 85-0179. This support is gratefully acknowledged. The specimens used in this test program were manufactured and supplied by the Lockheed-Georgia Company, Marietta, Georgia. The authors thank Mr. Timothy Moore for his assistance in performing the experiments.

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FIGURE CAPTIONS

Figure 1. The double cracked-lap-shear specimen

Figure 2. Fixture used to suppress mode I in a double cracked-lap-shear specimen.

Figure 3. Sublaminate description and coordinate system.

Figure 4. Dependence of P/F on the position of normal load application

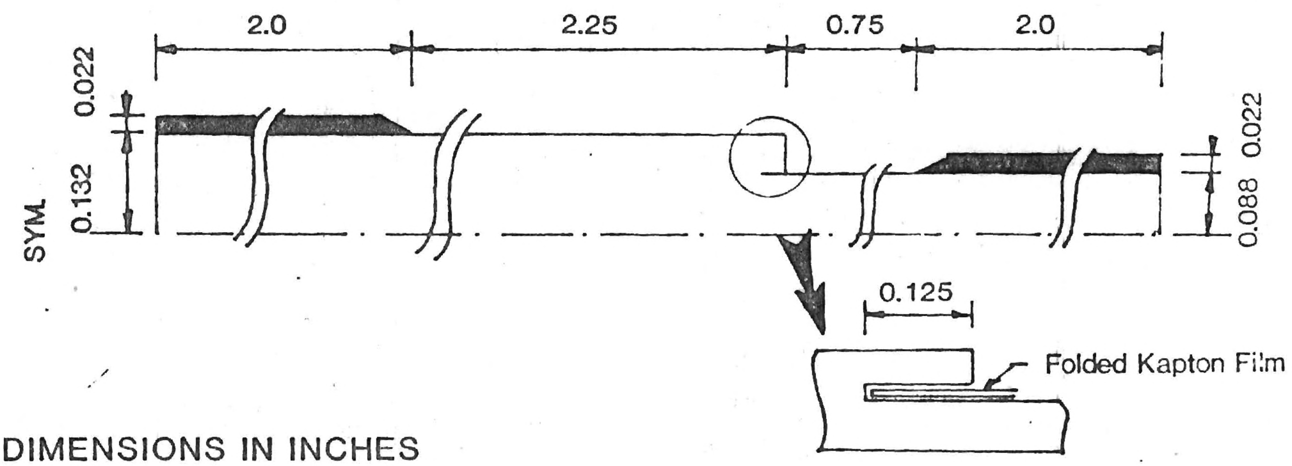
Figure 5. Comparison of crack growth resistance data with and without mode I suppression.

Figure 6. Effect of mode I suppression force on the crack initiation load.

Figure 7. Tension-Tension fatigue crack growth results with and without mode I suppression.

Figure 8. Compression-Compression fatigue crack growth results.

Figure 9. Notation and sign convention for the kth ply.



DIMENSIONS IN INCHES

(1in=25.4mm)

STRAP : (45/−45/0/90)s

LAP : (−45/45/0/90)2 s

Fig.1−The double cracked−lap−shear specimen.

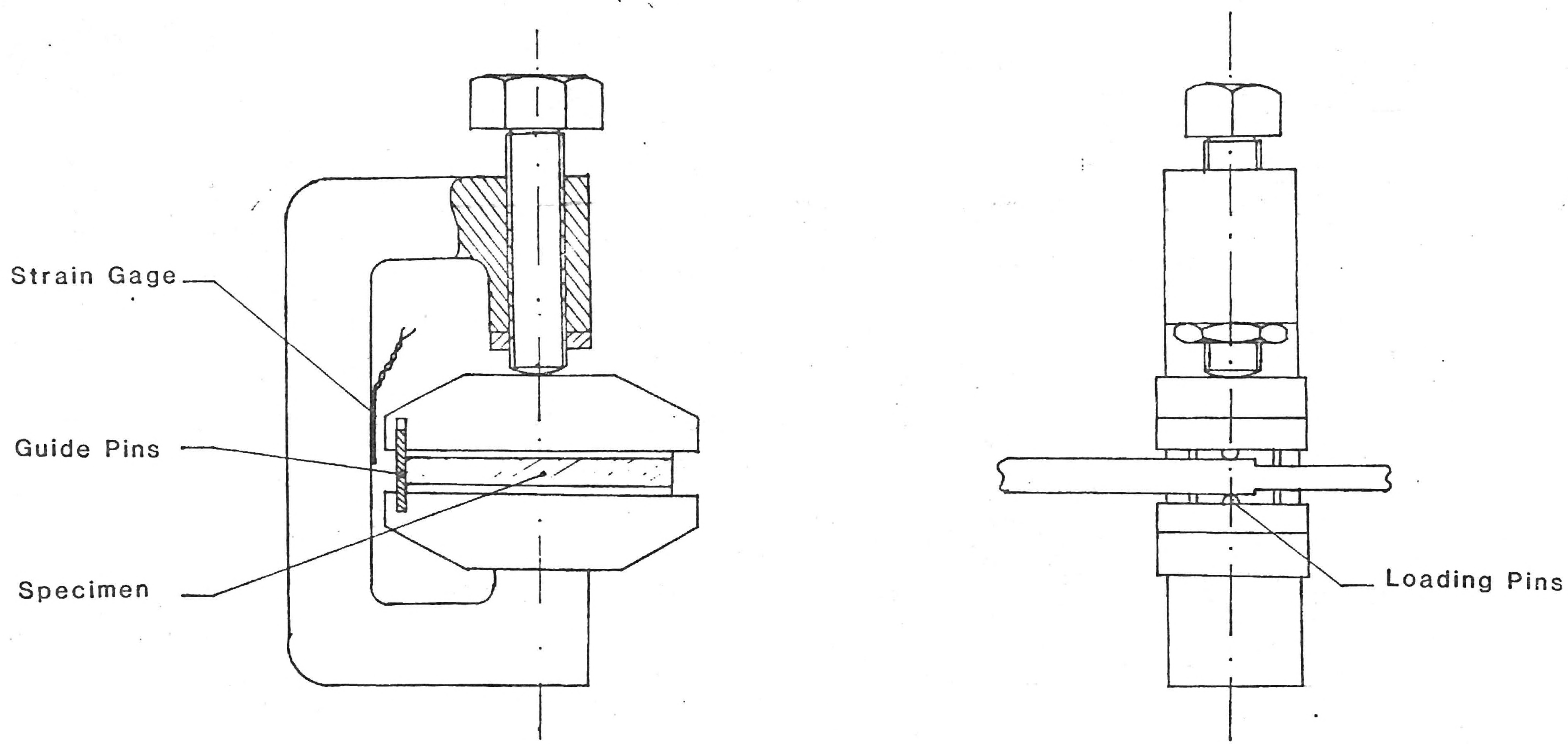


Fig.2-Fixture used to suppress mode I in a double cracked-lap-shear specimen

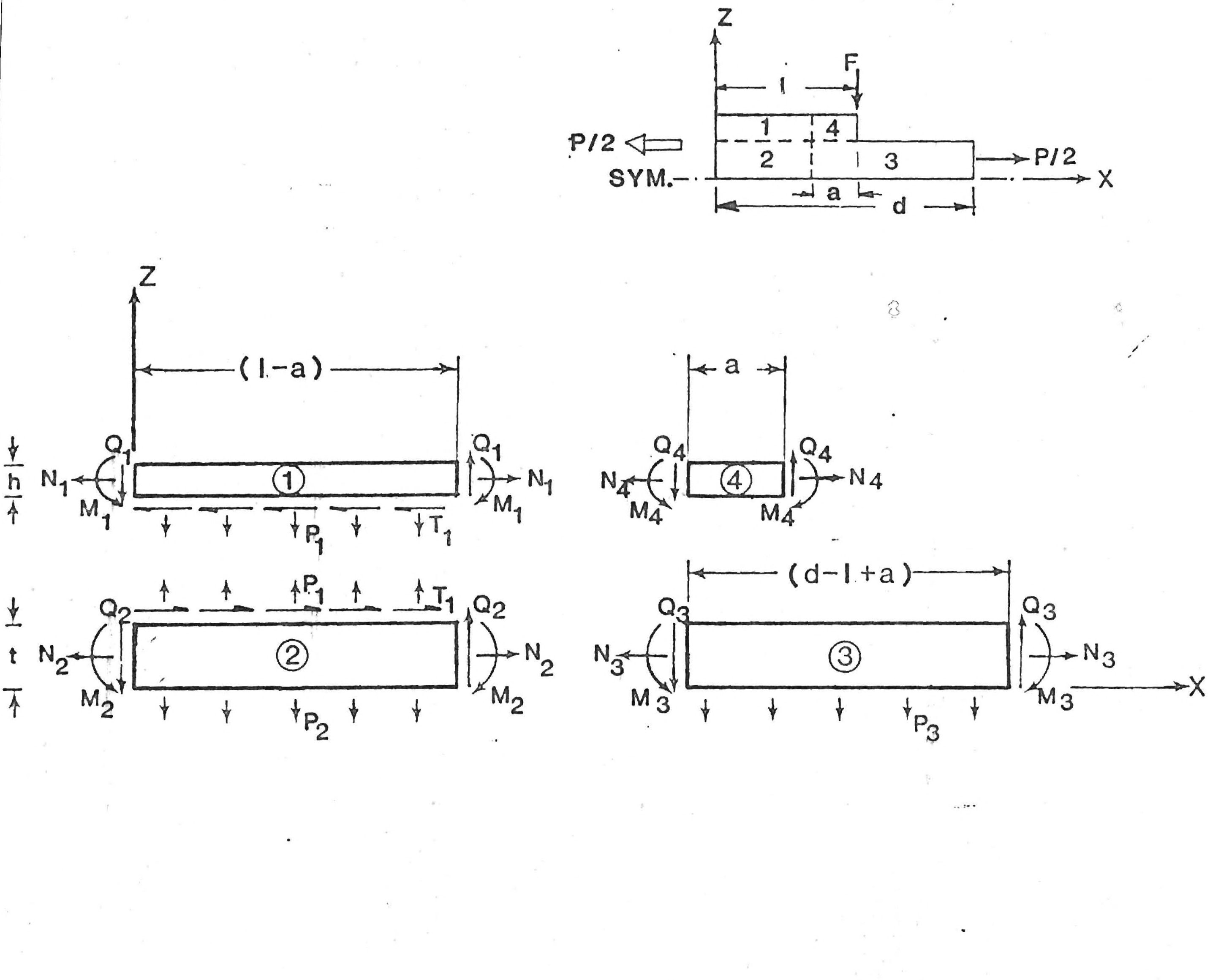


Fig.3- Sublaminate description and coordinate system

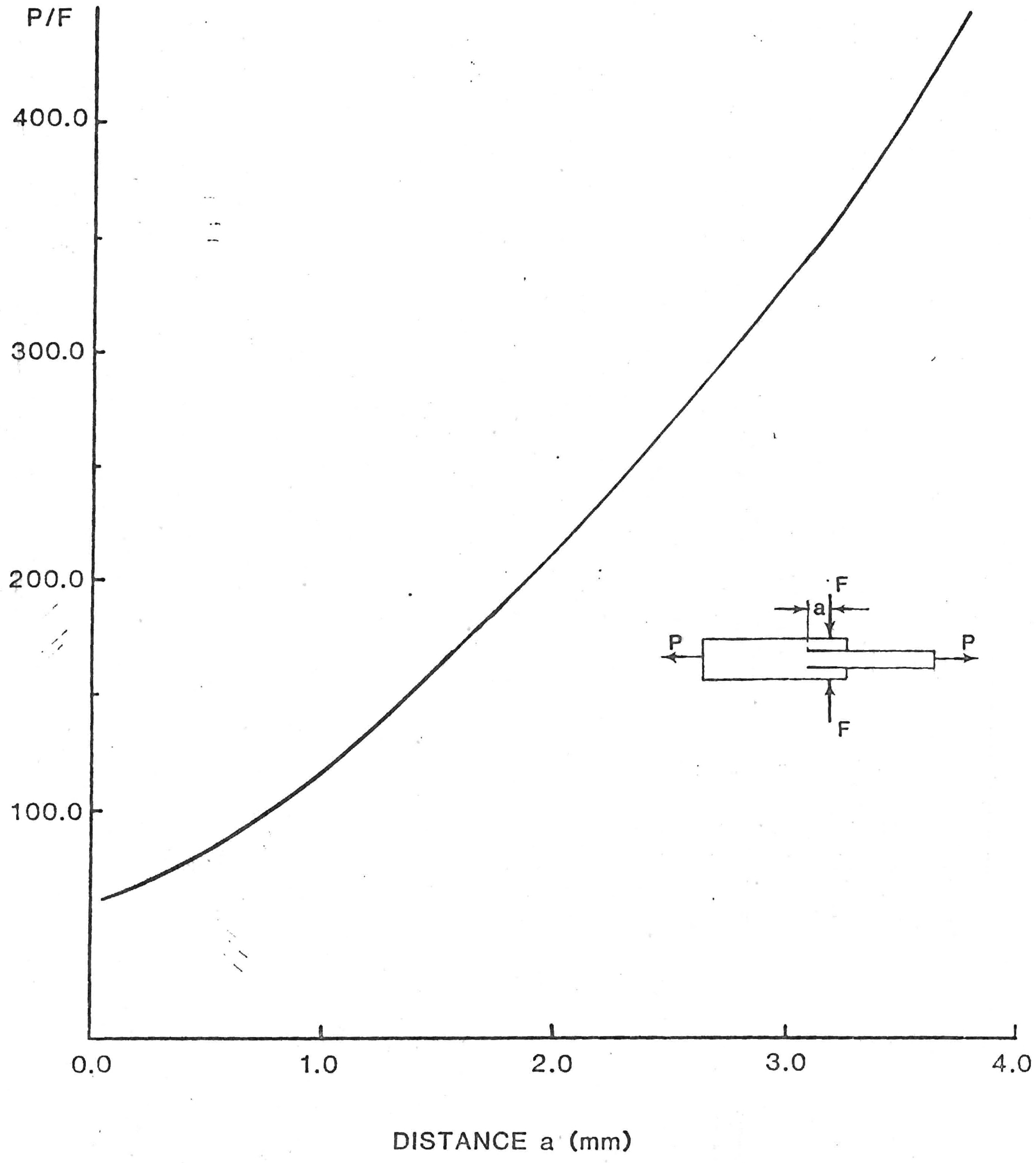


Fig.4-Dependence of P/F on the position of normal load application.

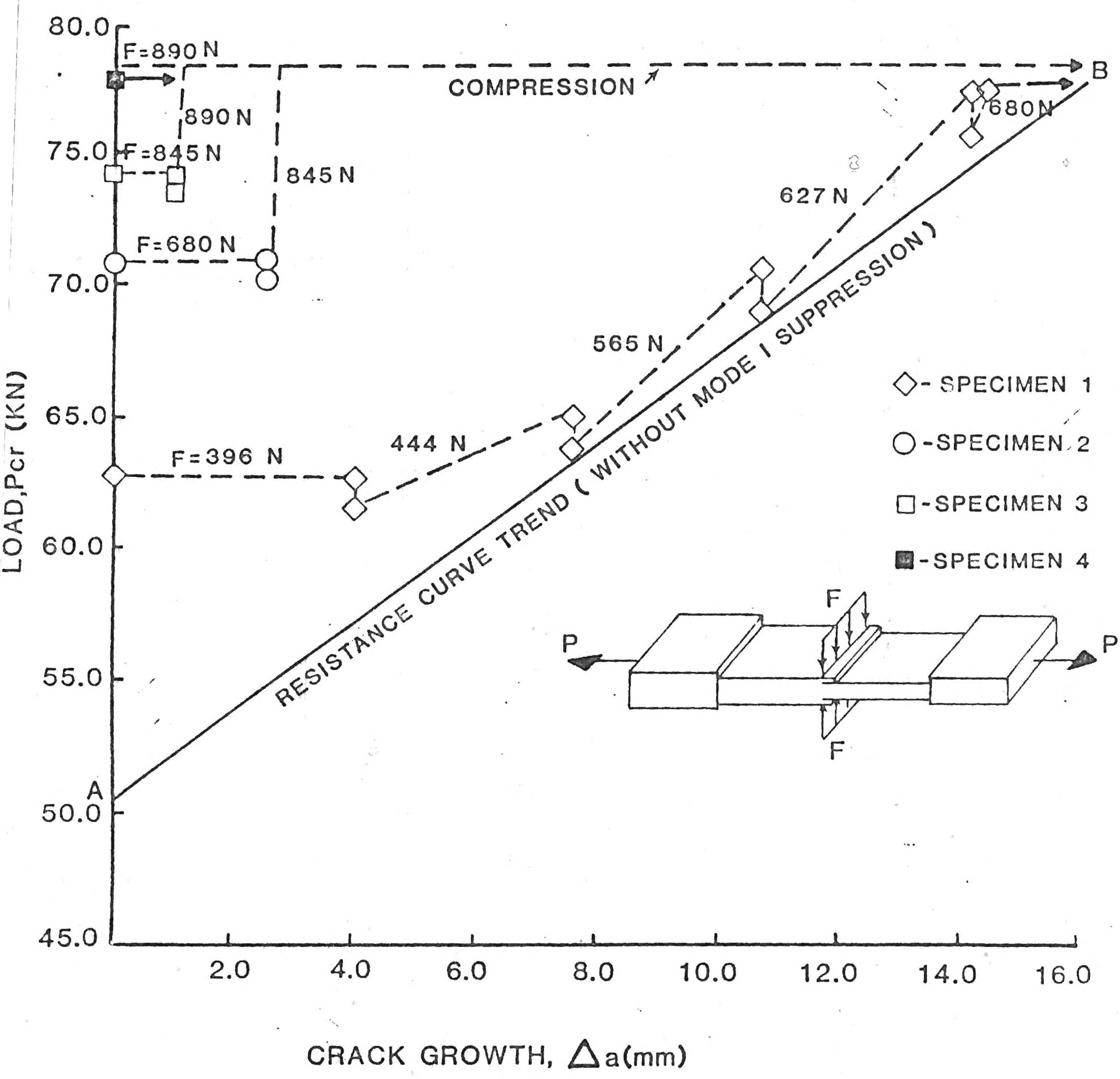


Fig.5- Comparison of crack growth resistance data with and without mode I suppression.

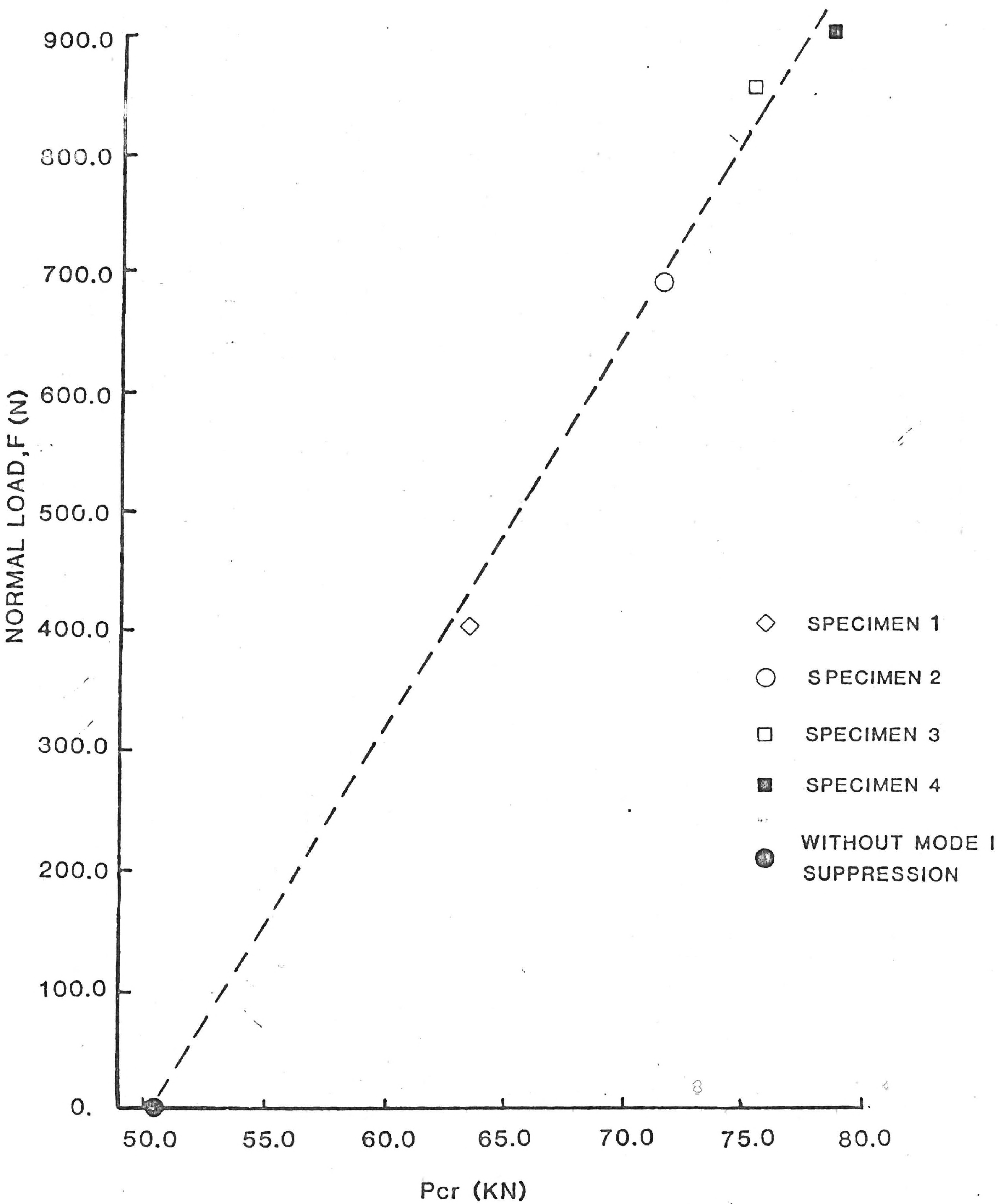


Fig.6– Effect of mode I suppression force on the crack initiation load.

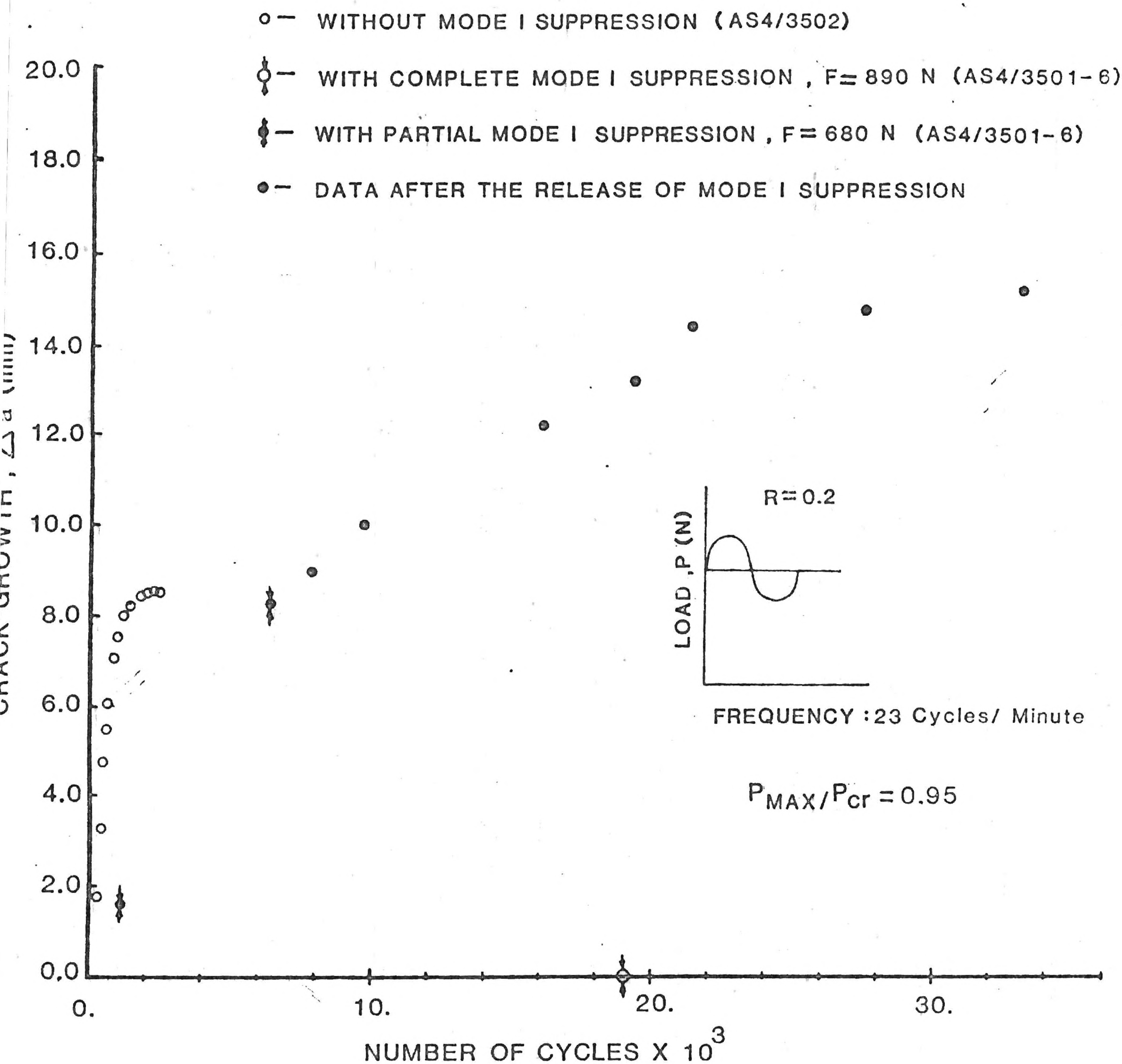


Fig.7- Tension-Tension fatigue crack growth results

with and without mode I suppression.

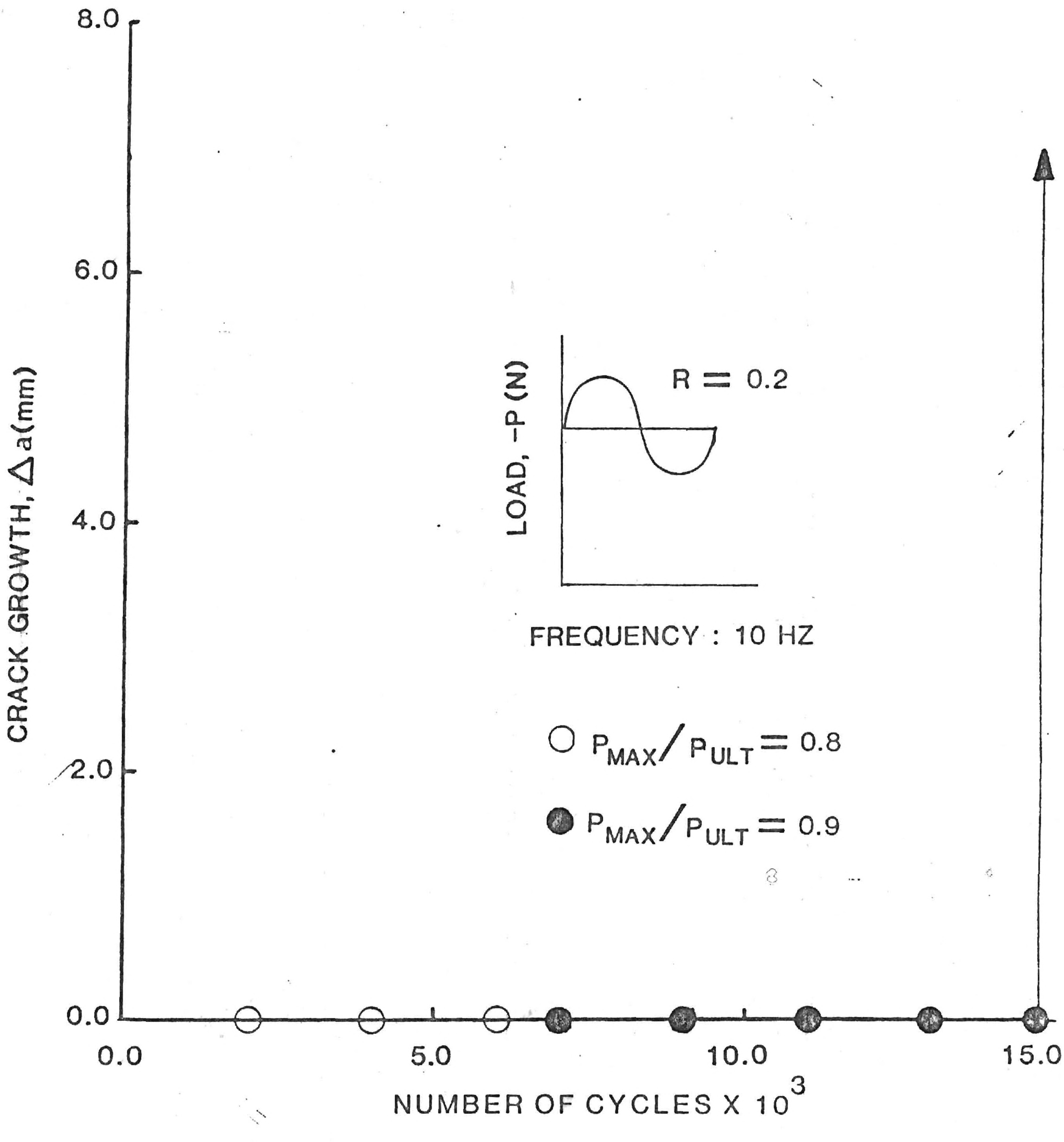


Fig.8– Compression–Compression fatigue crack growth results.

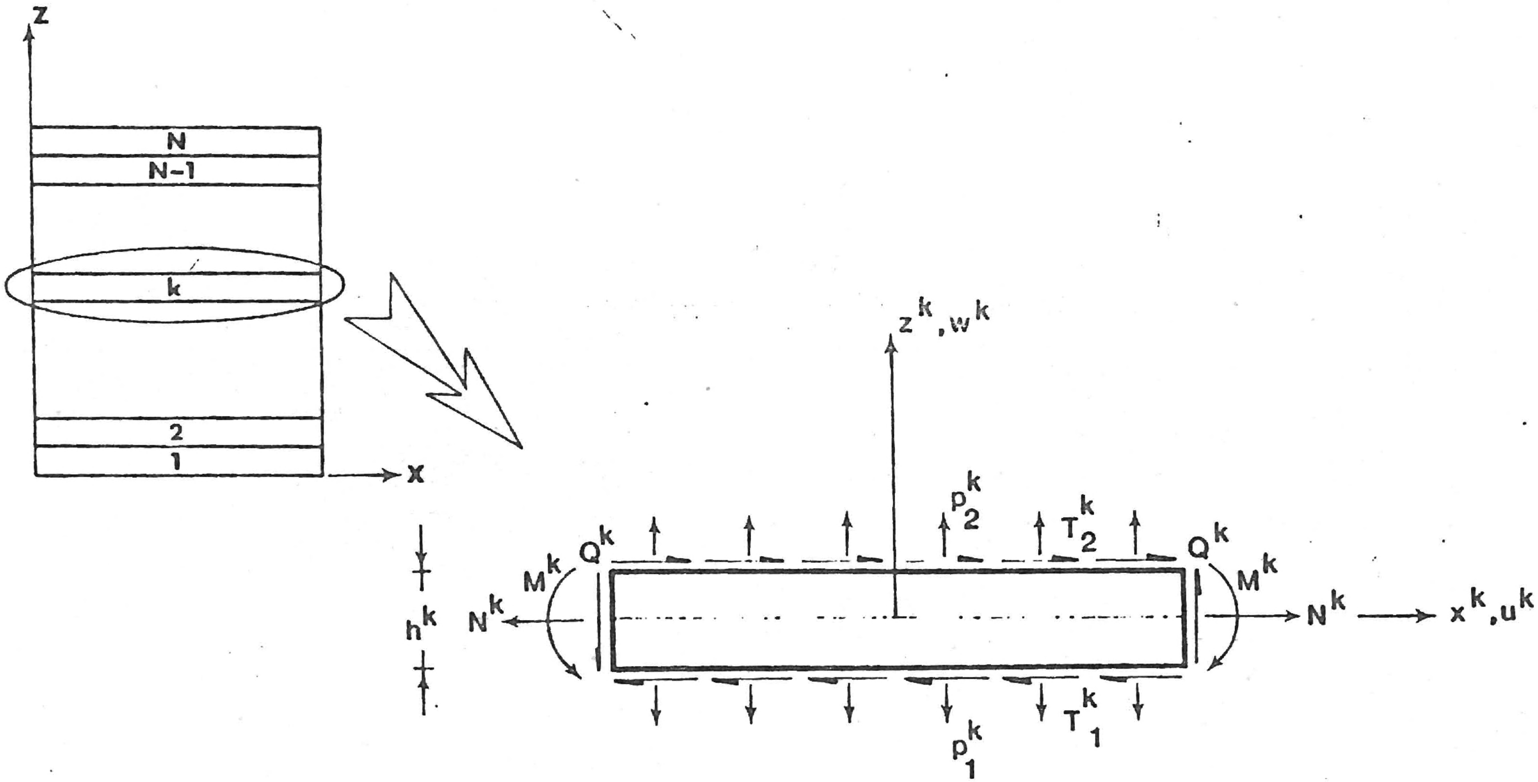


Fig.9- Notation and sign convention for the kth ply.

ACHIEVEMENT

SUBJECT: INTERLAMINAR FRACTURE OF
RESIN MATRIX COMPOSITES

- NEW TEST METHOD AND SPECIMEN CREATED
- EXPERIMENTS SHOW FUNDAMENTAL DIFFERENCE IN TENSION AND COMPRESSION BEHAVIOR
- TENSION
  - INITIAL STABLE CRACK GROWTH
  - TERMINAL, UNSTABLE FRACTURE
- COMPRESSION
  - UNSTABLE FRACTURE ONLY
  - CORRESPONDS TO UNSTABLE TENSION FRACTURE

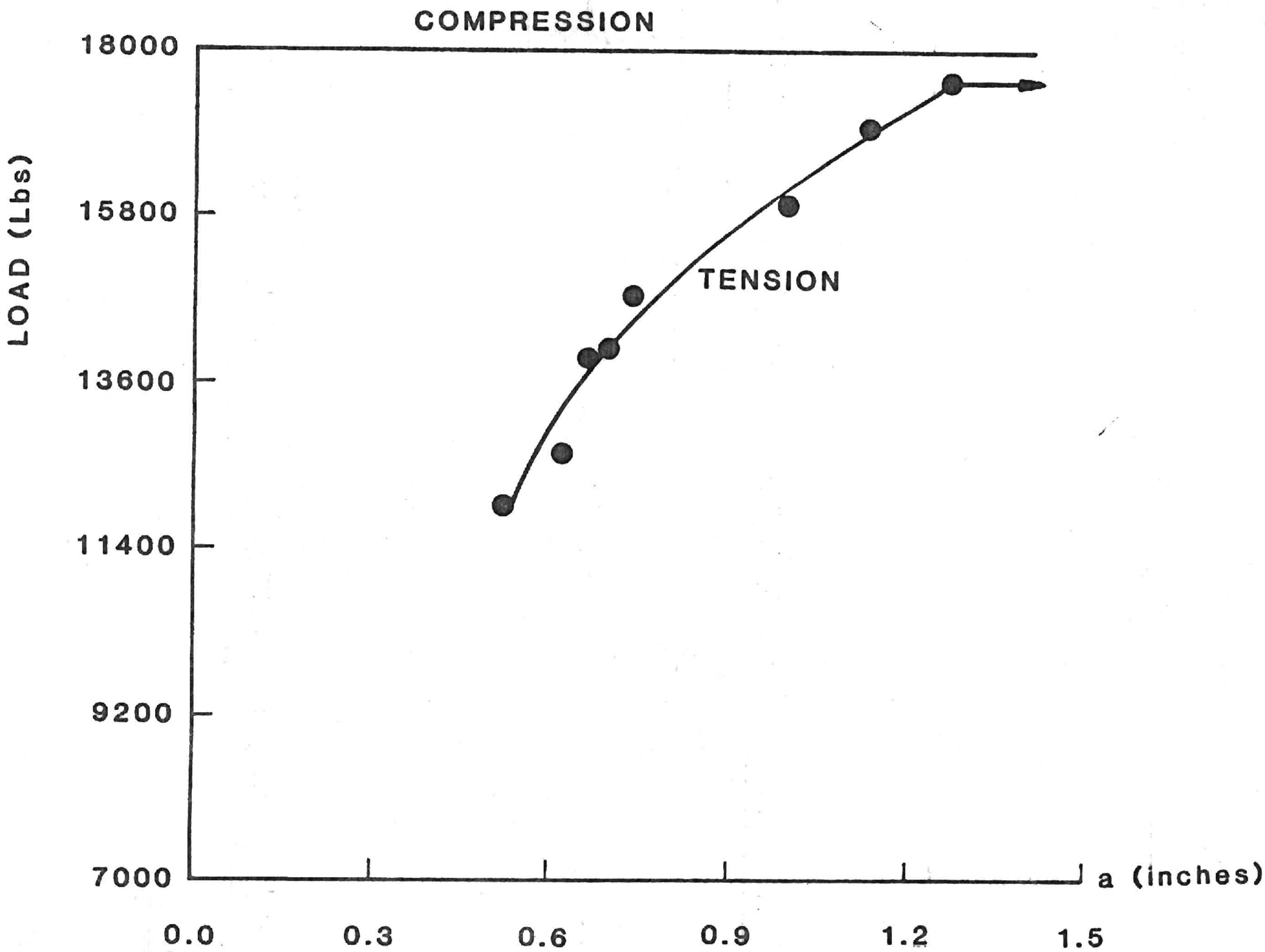


Figure 1. Tension/Compression Test Data

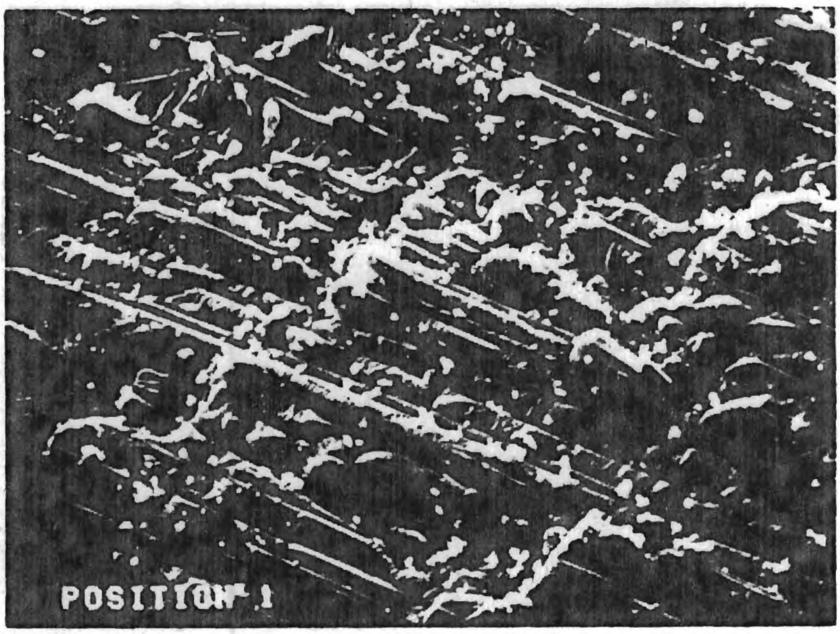


Figure 2. Photomicrograph of Fracture Surface – Original Magnification X 300 – Stable Crack Growth Region

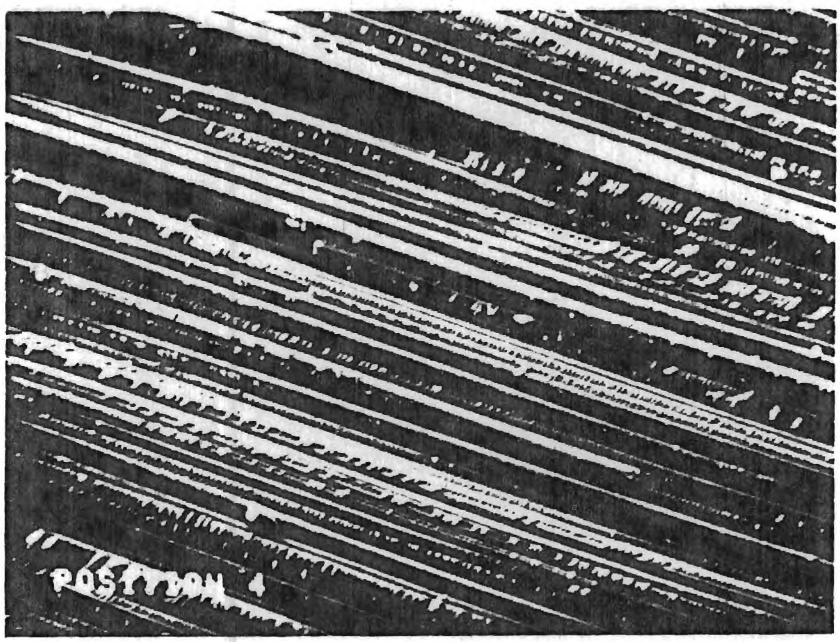


Figure 3. Photomicrograph of Fracture Surface – Original Magnification X 300 – Unstable Crack Growth Region

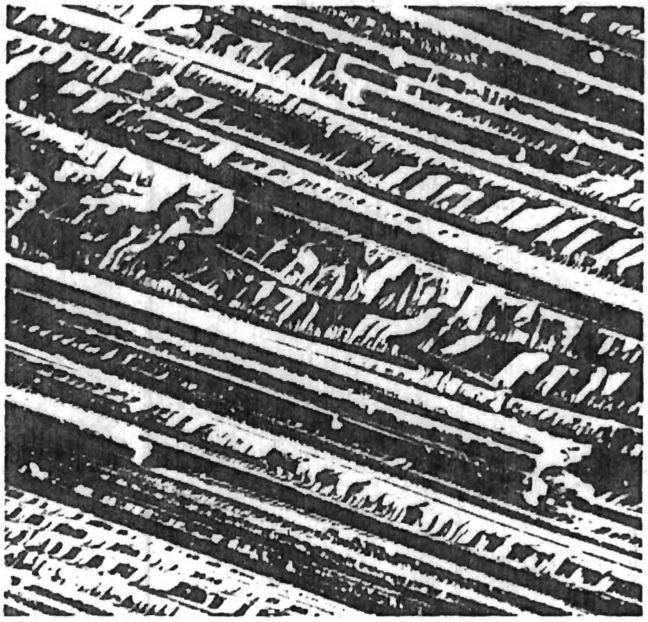


Figure 4. Photomicrograph of Fracture Surface - Original Magnification X 500 - Unstable Crack Growth Region - Tension

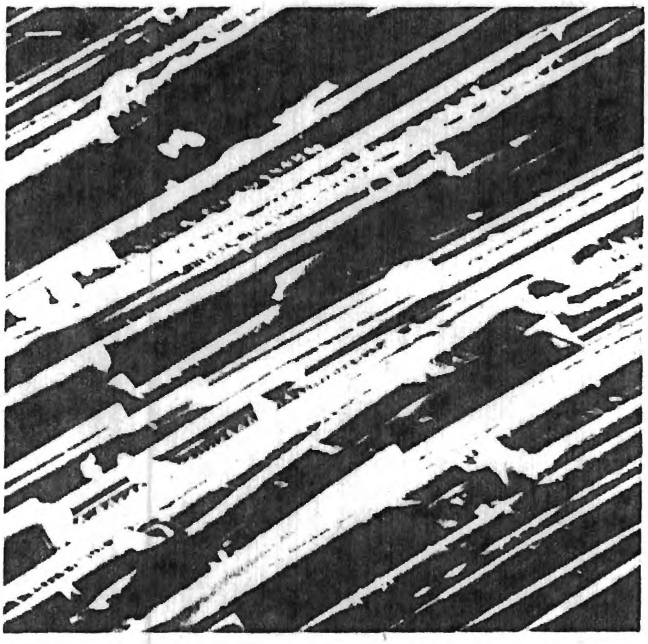


Figure 5. Photomicrograph of Fracture Surface - Original Magnification X 500 - Unstable Crack Growth Region - Compression

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FIELD	GROUP	SUB. GR.	DELAMINATION; DAMAGE IN COMPOSITES; DAMAGE TOLERANCE; COMPOSITE MATERIALS

19. ABSTRACT (Continue on reverse if necessary and identify by block number)

This final report summarizes the objectives and accomplishments of research on sublaminate damage mechanisms in composite structures. The work was performed during the period 15 April 1985 - 14 April 1987. It may be separated into three elements. The first is interlaminar fracture analysis methodology development. Existing methods for predicting strain energy release rate compoments have been assessed, and two new alternative approaches have been developed and illustrated.

The second is the creation of a phenomenologically based criterion for damage tolerance analysis of composite structures. An equivalent damage parameter and a relevant compliance measure are determined from experiments. This methodology has been validated by seven independent experiments.

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The third is the creation of a sublaminate damage analysis methodology which ermits the understanding and prediction of the interaction between damage mechanisms. t is applied to tensile failure of a double cracked-lap-shear specimen where there is nteraction between delamination and matrix microcracking. analytical predictions are n good agreement with experimental findings for this configuration.

SUBLAMINATE DAMAGE MECHANISMS IN COMPOSITE STRUCTURES

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School of Aerospace Engineering
Georgia Institute of Technology
Atlanta, Georgia 30332-0150

Final Scientific Report
15 April 1985 - 14 April 1987
AFOSR Grant No. 85-0179

INTRODUCTION

The work described herein was performed at the School of Aerospace Engineering, Georgia Institute of Technology during the period 15 April 1985 - 14 April 1987. Professor Lawrence W. Rehfield was the Principal Investigator.

This research concerns the mechanisms of damage that occur in composite structures on the sublaminate scale - - - that is the scale of individual plies or groups of plies. The objectives have been to (1) obtain a fundamental understanding of the damage processes from a structural point of view, (2) model them appropriately so that trends or events can be predicted and key parameters identified and (3) use the predictions to provide guidelines, which are hopefully easy to understand and apply, to designers on detailed design issues which influence stress raisers that are common in composite structures.

The usual approach to dealing with localized phenomena is large scale numerical simulation and analysis, mostly by general purpose finite element codes. This approach is often supplemented by a "build and test" demonstration, or series of demonstrations if repeated failures are encountered. While such approaches are often costly and inefficient, their major drawback is that fundamental principles are not discovered which provide the means to produce better results. Furthermore, the steps must be repeated all over again the next time a similar situation arises.

In contrast to the above approach, we have pursued fundamental objectives which have been met in the main. In particular and, to our knowledge, for the first time, an analytical framework has been created which permits the interaction of damage mechanisms in composite laminates to be predicted and, more importantly, understood. This is believed to be a major contribution. Furthermore, correlation with carefully performed experiments provide convincing evidence of the validity of the analytical methodology.

The second significant contribution is creating a means of relating complex forms of damage directly to load carrying capacity or residual strength on a reliable, physically meaningful basis. Many have believed that observed characteristic damage states commonly found in composites due to fatigue, static loading and low energy impact are so complex that they defy analytical prediction. This may be true if a traditional atomistic, detailed mechanical model is developed and analyzed. Predicting every individual event in order to grasp the overall reality may not produce the desired insight and understanding to control the process in an engineering sense. It is certainly not economical or efficient. A concept which we call "equivalent damage" permits the detailed, atomistic steps to be avoided in making practical predictions on the effects of damage in composites. This approach is supported by data from seven independent experiments.

SUMMARY OF ACCOMPLISHMENTS

Foundation Provided by Previous Work

The present research had its origin in the development and application of new structural models for composites under three previous AFOSR grants, 81-0056, 82-0080 and 83-0056. This modeling technology permitted prediction of interlaminar stresses in composite laminates and strain energy release rates for delamination prediction by elementary means. This is an enabling technology.

Delamination poses some unique problems for the analyst. In practical composite structures built up from many plies, each interply surface is a potential delamination site. Consequently, a complete analysis must be conducted for each interply surface by assuming the crack occurs in that plane. The fact that numerous repetitive analyses must be conducted is of great practical importance. Issues such as the efficiency of the analysis method and the elapsed time required impact both cost and schedule in a major way. This situation provided the motivation for development of this new approach.

Computer codes based upon this work are now used regularly at the NASA Langley Research Center, U.S. Army Aerostructures Directorate, Air Force Materials Laboratory and Bell Helicopter Textron, Inc.

Earlier work, which was completed and published during the current grant period and which contributed to the present research, appears in Accomplishments 1, 2 and 5. The purpose of citing these accomplishments is two-fold: (1) they provide the background that has proved so essential to our modeling efforts, which are key to the present research and (2) effort was expended during the present grant period in order to complete them and bring them to publication.

Overview of the Research

The research program can be separated into three elements. The first is the completion of background work on interlaminar fracture analysis which had its origin in the research conducted under the earlier grant 83-0056. This is a key element as interlaminar fracture or delamination is a very common damage mode in laminated composite structures. It is prevalent because interlaminar planes have minimum fracture toughness.

These planes are not reinforced by fibers. A complete account of this work appears in Accomplishments 3, 4 and 6.

The second is the development of a new approach to damage tolerance analysis and testing of composite structures. It has its origin in experimental observations, and it provides a means of relating damage size to failure. Thus, it is intended to serve the same purpose as fracture mechanics does for metals. This new approach is presented in Appendix I.

The third is the development of the analysis methodology that permits the understanding and prediction of the interaction of sublaminate damage mechanisms. It is presented and applied in Appendix II. Also, it will be presented as indicated in Accomplishment 15.

Each of these three elements will be considered in the following sections.

<u>Interlaminar Fracture Analysis Methodology</u>

Interlaminar fracture in composite laminates can be predicted by an energy release rate analysis based upon fracture mechanics. Several fracture laws for composite structures are expressed in terms of the energy release rate components. Therefore, an accurate knowledge of their values is essential to design against fracture. An extensive investigation of two major approaches which utilize finite element analysis to obtain the energy release rate components has been performed[3,6]. These approaches are the crack closure technique and the virtual crack extension method. A dependence of the energy release rate components on the mesh and crack extension size is identified. A new approach which utilizes results obtained by the crack closure technique and the virtual crack extension method is presented. This approach leads to improved results which are independent of the mesh and crack extension size. Improved estimates for a benchmark mode I behavior have been found and compared to an exact solution to prove the efficiency of this approach. This method is also applied successfully to a mixed-mode configuration.

Another new method[3], the coupled strain energy method, has been developed for calculating the energy release rate (ERR) components. This approach exploits the superposition of an auxiliary equilibrium state to the mixed mode situation under consideration. Finally, separate ERR components are obtained in terms of the auxiliary solutions and the coupled strain energy. These reliable predictions can be utilized with confidence in appropriate failure laws for composite materials systems.

This work has provided a base of knowledge that has proven of enormous value in the development of the analysis methodology for sublaminate damage. It is pioneering in nature and provides new methods of ERR analysis and the conceptual foundation for the methods.

Damage Tolerance Analysis

A new, promising approach to damage tolerance analysis for composite structures has been created. A damage law relating equivalent damage size and failure has been established which has been validated by seven independent experiments on several specimen configurations. Four generic configurations have been studied. They are:

1. Double cracked-lap-shear (discrete ply dropoff),
2. Taper by dropping or terminating plies,
3. Damaged imbedded plies, and
4. Through thickness holes.

All four produce high interlaminar stresses and delamination accompanied by other forms of damage. These specimens progressively introduce fiber controlled damage in addition to delamination and matrix microcracking.

An equivalent damage parameter, which is monotonically related to the extent of damage, and a relevant compliance measure must be determined from experiments. These are input to the damage law. Complete details are presented in Appendix I.

While this approach is phenomenological in nature, it provides a practical means of assessing the effects of complex states of damage on residual strength or failure. At present the only alternative is empiricism.

Sublaminate Damage Analysis

The origin of this work is a series of delamination experiments conducted on double cracked-lap-shear specimens in tension. After the onset of delamination, stable crack growth occurred --- increasing loads were required to extend the delamination crack. There was no known physical mechanism that explained this behavior as the graphite-epoxy material system was quite brittle. After considerable investigation, the interaction of two damage modes --- delamination and matrix microcracking -- was found to provide a rational explanation that is in concert with analysis predictions and physical evidence. To the authors' knowledge, this is the first successful analysis of the interaction between delamination and matrix microcracking.

A key assumption was used in creating the analysis methodology --- matrix microcracking is predicted by strain level only. This is valid for damage characteristic dimensions greater than a ply thickness. For damage on a smaller scale, fracture mechanics concepts and means for detecting sub-ply microcracks is required. For most engineering applications, strain level predictions are quite satisfactory.

A complete account of this work appears in Appendix II. An application to a double cracked-lap-shear specimen in tension provides a convenient test case where experimental data exists.

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4. Rehfield, L.W., Armanios, E.A. and Weinstein, F., "Analytical Modeling of Interlaminar Fracture in Laminated Composites," Composites '86: Recent Advances in Japan and the United States, Proceedings of the Third Japan-U.S. Conference on Composite Materials, K. Kawata, S. Umekawa, and A. Kobayashi Eds., 1986 pp. 331-340.

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Publications Pending

6. Weinstein, F., Armanios, E.A. and Rehfield, L.W., "An Assessment of Methods for Computing Energy Release Rate Components for Interlaminar Fracture," ASTM Second Symposium on Composite Materials: Fatigue and Fracture, Cincinnati, OH, 27-28 April 1987. To appear in an ASTM STP volume.

Presentations

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Presentations (continued):

10. Weinstein, F., Armanios, E.A. and Rehfield, L.W., "An Assessment of Methods for Computing Energy Release Rate Components for Interlaminar Fracture," presented at the ASTM Ninth Symposium on Composite Materials: Fatigue and Fracture, Cincinnati, Ohio, 26 April 1987.

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b. Fracture and Fatigue Research Laboratory, Georgia Institute of Technology, Atlanta, Georgia, 20 January, 1987.

c. Centennial Colloquium Series in Structural Mechanics, School of Engineering Science and Mechanics, Georgia Institute of Technology, 30 January, 1987.

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APPENDIX I

"DAMAGE TOLERANCE ANALYSIS AND TESTING OF COMPOSITE AIRCRAFT STRUCTURES"

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INTRODUCTION

Damage tolerance for composite aircraft structures has an empirical base. There is no unifying framework such as fracture mechanics for metal structures. This is due primarily to the complexity of damage modes in composites which includes contributions with different length scales.

There is a pressing need for a damage tolerance analysis methodology for composites. It is needed not only for design but for supportability. Rational maintenance/repair decisions affect both readiness and life cycle costs. At present, empirical data are mainly used. There are also some current efforts aimed at modeling micromechanisms observed in foreign object impact which utilize test data as input.

The objective of this proposed work is to investigate whether a newly discovered framework and approach can be applied to foreign object impact damage. This new approach is briefly described in the following.

FOUNDATION OF A NEW APPROACH

There are two types of information that are needed for a damage tolerance analysis. The first is a quantitative assessment of the damage in the structure. The second is a relationship between failure and damage size or extent. Damage assessment is straightforward in metal structures --- crack area --- and most cracks reach the exterior surfaces where they can be detected. Fracture mechanics provides the failure criterion in this case.

If residual strength is adequate, that is, there is no immediate static failure for the given damage, the residual lifetime is needed in order to establish rational inspection procedures and repair decisions. Life predictions require additional data. These data characterize damage growth in fatigue and are usually in the form of fatigue damage growth as a function of stress intensity for metals.

The situation for composites is not nearly so neat and well established. Difficulties begin with the complex modes of damage which are difficult to describe and quantify. With the exception of pure interlaminar fracture or delamination, fracture mechanics cannot be used directly. This is because damage growth is not self-similar and mode changes and mode interactions are common in practical laminates.

The new approach presented here is a direct result of experimental observations. A generic situation is depicted in Figure 1. Almost continuous damage, a progressive form of damage that develops as a series of sequential damage increments, is indicated; this is commonly found when delamination and matrix cracking are major damage modes. Also shown is a compliance plot which illustrates compliance change with damage size. Note that a compliance value corresponding to failure is approached.

The compliance-damage relationship indicated in Figure 1 can be modeled by the damage growth law presented in Figure 2. This law serves the same purpose that fracture mechanics does for metals that are damaged by cracking. It is a two parameter model. The parameter a_o characterizes the integrated mechanical effect of initial microdamage. The parameter C_F is the compliance corresponding to failure.

Figure 1

ALMOST CONTINUOUS DAMAGE

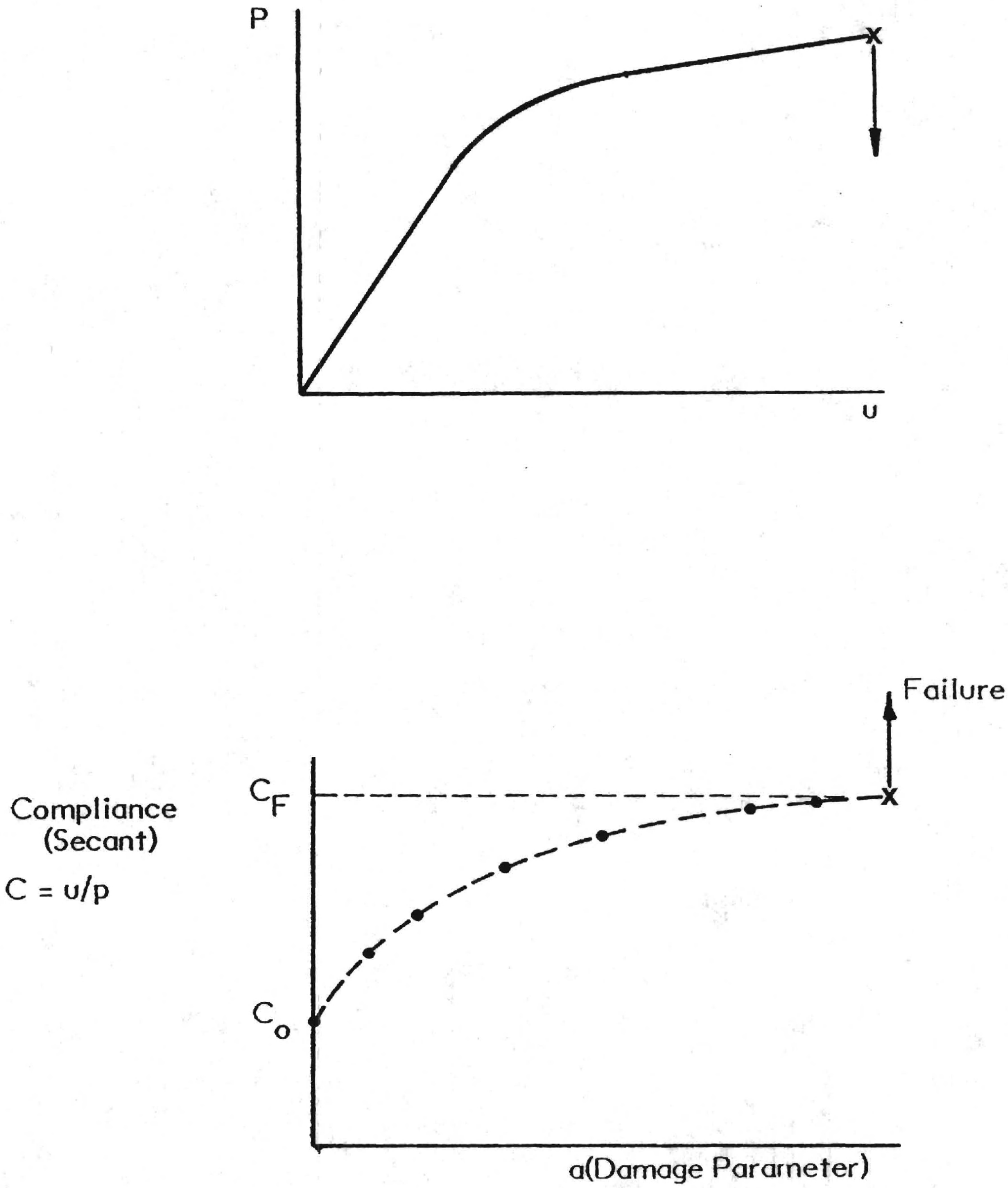


Figure 2

DAMAGE GROWTH LAW

C-a CURVE IS APPROXIMATELY HYPERBOLIC

$$a = a_o \frac{(C - C_o)}{(C_F - C)}$$

$$\text{TOTAL DAMAGE} = a_o + a$$

a_o, CHARACTERIZES INITIAL DAMAGE

a, CHARACTERIZES NEW DAMAGE GROWTH

If the damage growth represents the behavior of the structure, experimental data may be plotted in the particularly convenient manner shown in Figure 3. A linear "Damage Growth Plot" can be constructed. The intercept with the compliance axis corresponds to the failure compliance. The slope of the line is related to the initial damage size and can be used to quantify it.

This approach permits, therefore, failure to be predicted by extrapolation of subcritically obtained data. While this is not a nondestructive test in a strict sense, testing to failure is not required. The ability to quantify the initial damage by test facilitates the establishment of typical values which can be used in damage tolerant design procedures.

Our methodology was to hypothesize the form of the damage growth law in Figure 2 and to attempt to validate it for a number of generic test bed structures. A total of seven distinct experiments have been conducted on six different configurations. The generic configurations were designed to have a strong delamination damage component. Delamination area was the parameter chosen to characterize the damage. The six specimens are shown in Figures 4-9. A summary of the test results and subcritical data analysis predictions are given in Figure 10.

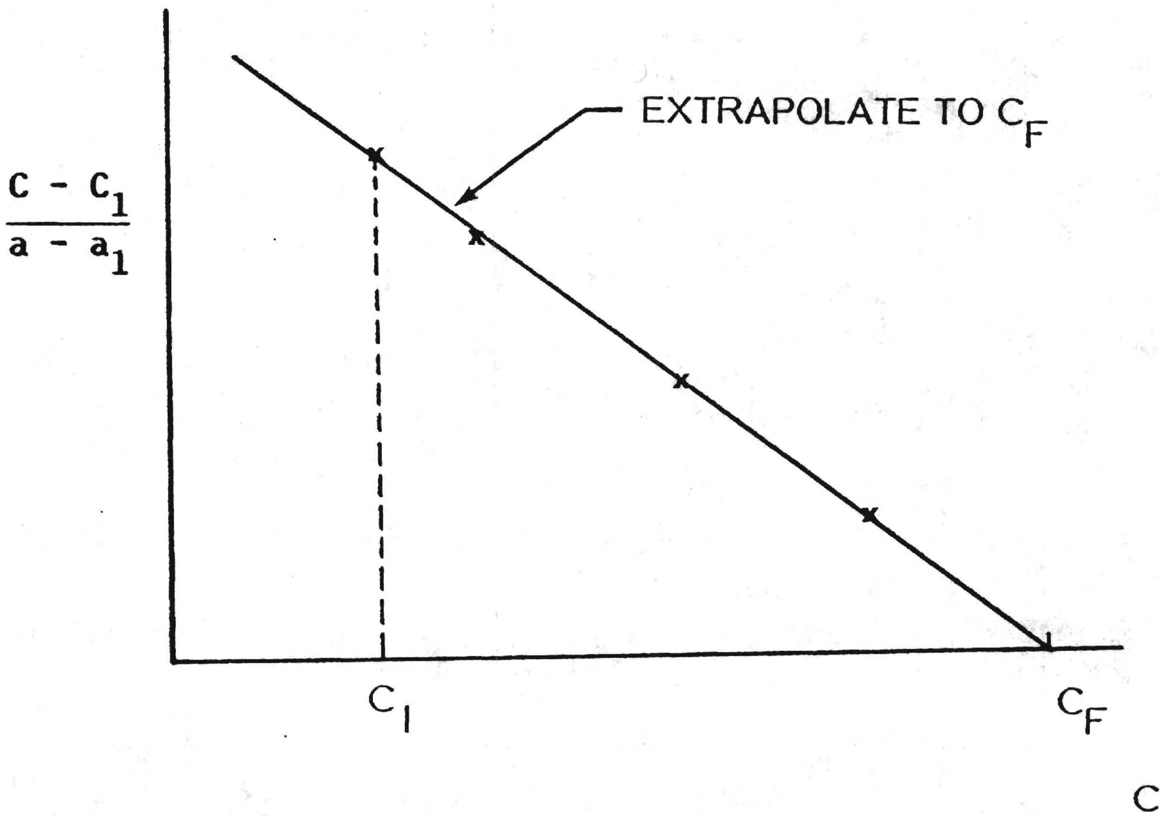
The proposed approach clearly works well for the test bed structures under tensile loading. This establishes a base of favorable experience that justifies pursuing the approach further. Note that the approach is phenomenological in nature. A detailed understanding of micromechanisms and modeling of them is not required.

Figure 3

<u>DAMAGE GROWTH LAW PERMITS</u>

<u>"STIFFNESS PLOT" TO BE USED</u>

$$(C - C_1)/(a - a_1) = \left[(C_F - C_1)/a_o C_F\right](C_F - C)$$



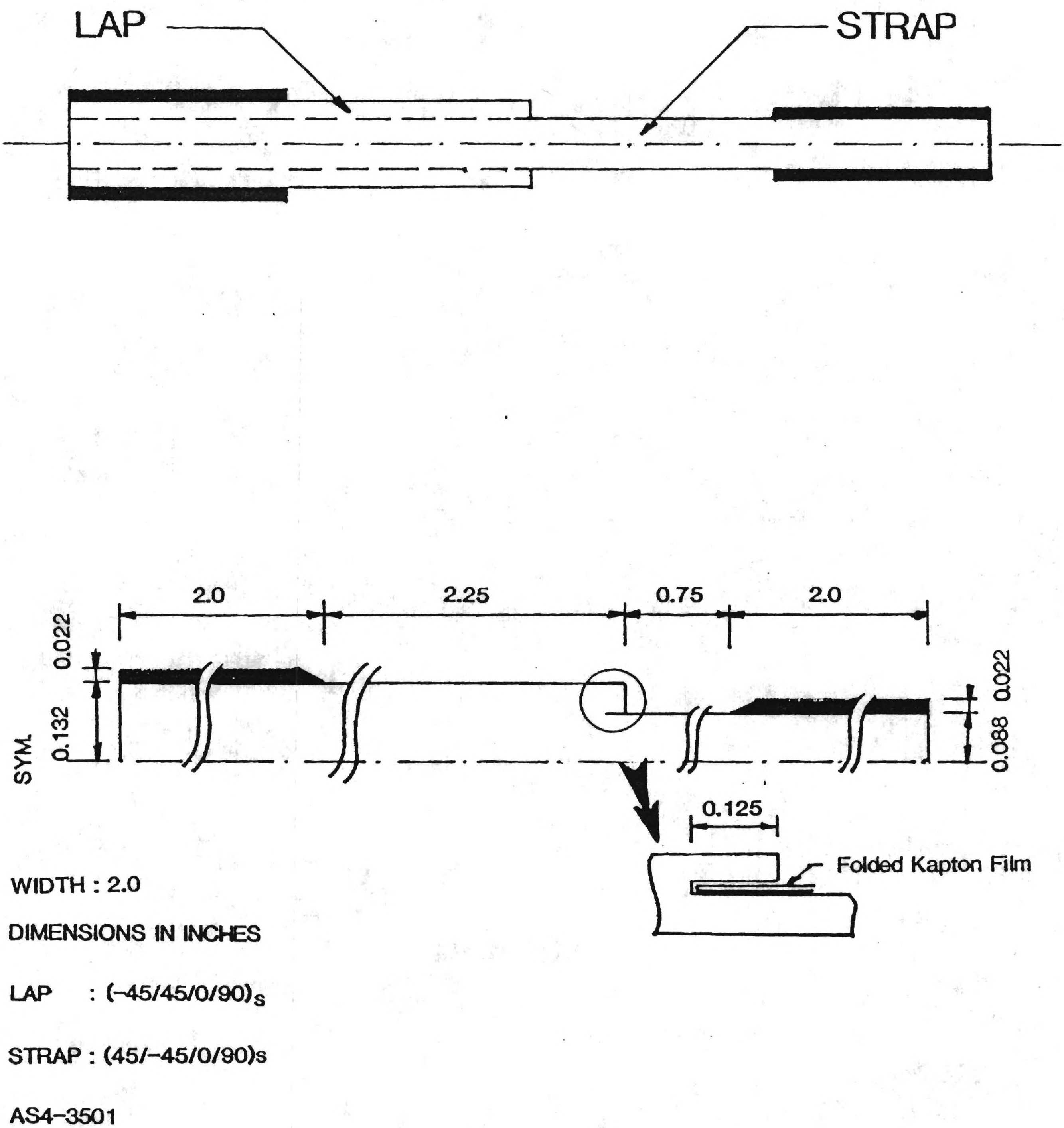
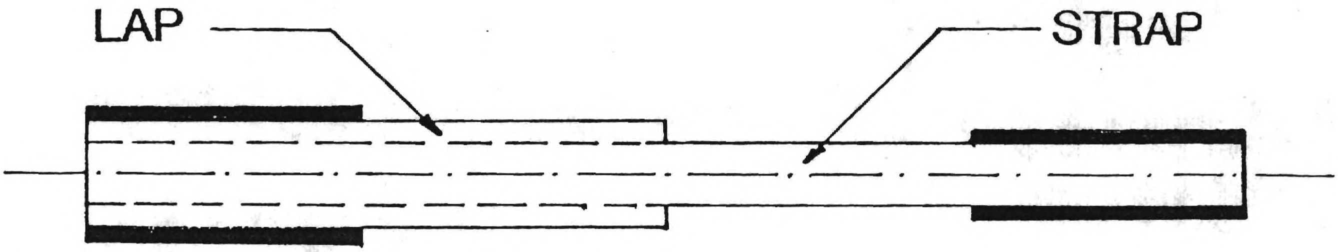
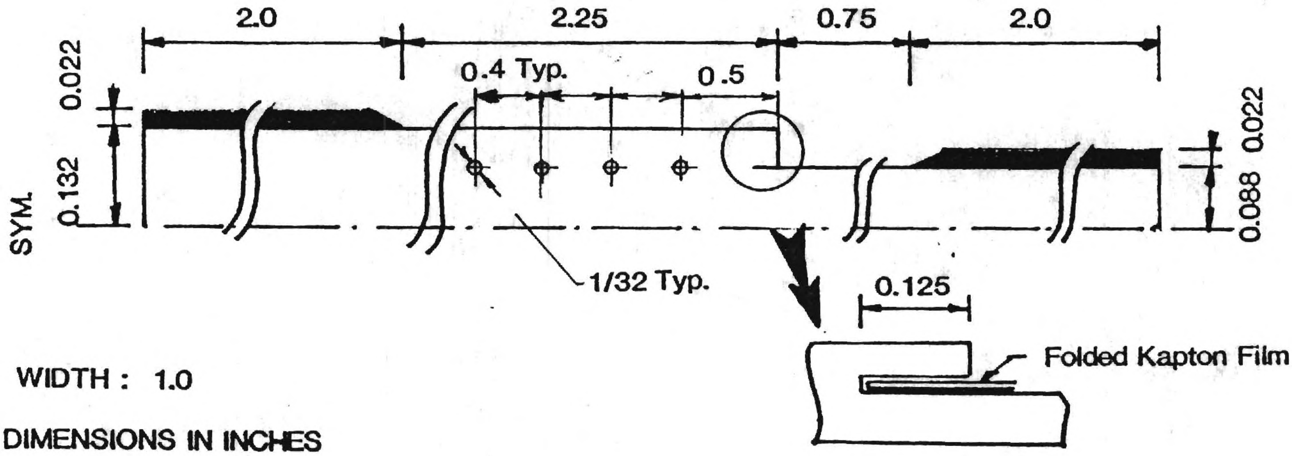


Figure 4

The Double Cracked-lap-shear Specimen





WIDTH : 1.0

DIMENSIONS IN INCHES

LAP : $(-45/45/0/90)_s$

STRAP : (45/−45/0/90)s

AS4−3501

Figure 5

The Double Cracked-lap-shear Specimen
With Internal Ply Damage :
0.4 IN Lateral Hole Spacing, 1/32 IN. DIA.

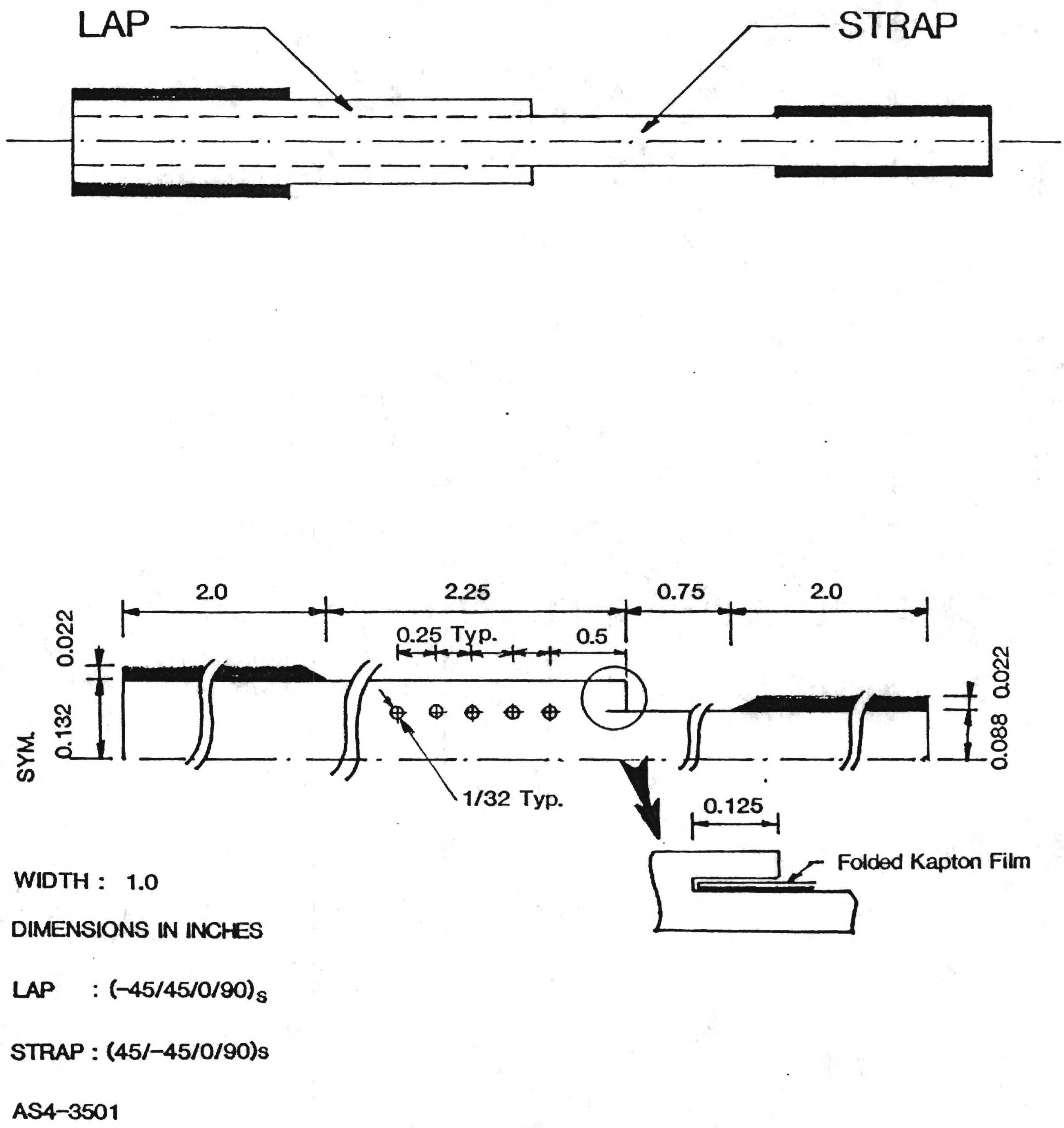


Figure 6

The Double Cracked-lap-shear Specimen
With Internal Ply Damage :
0.25 IN. Lateral Hole Spacing ,1/32 IN. DIA.

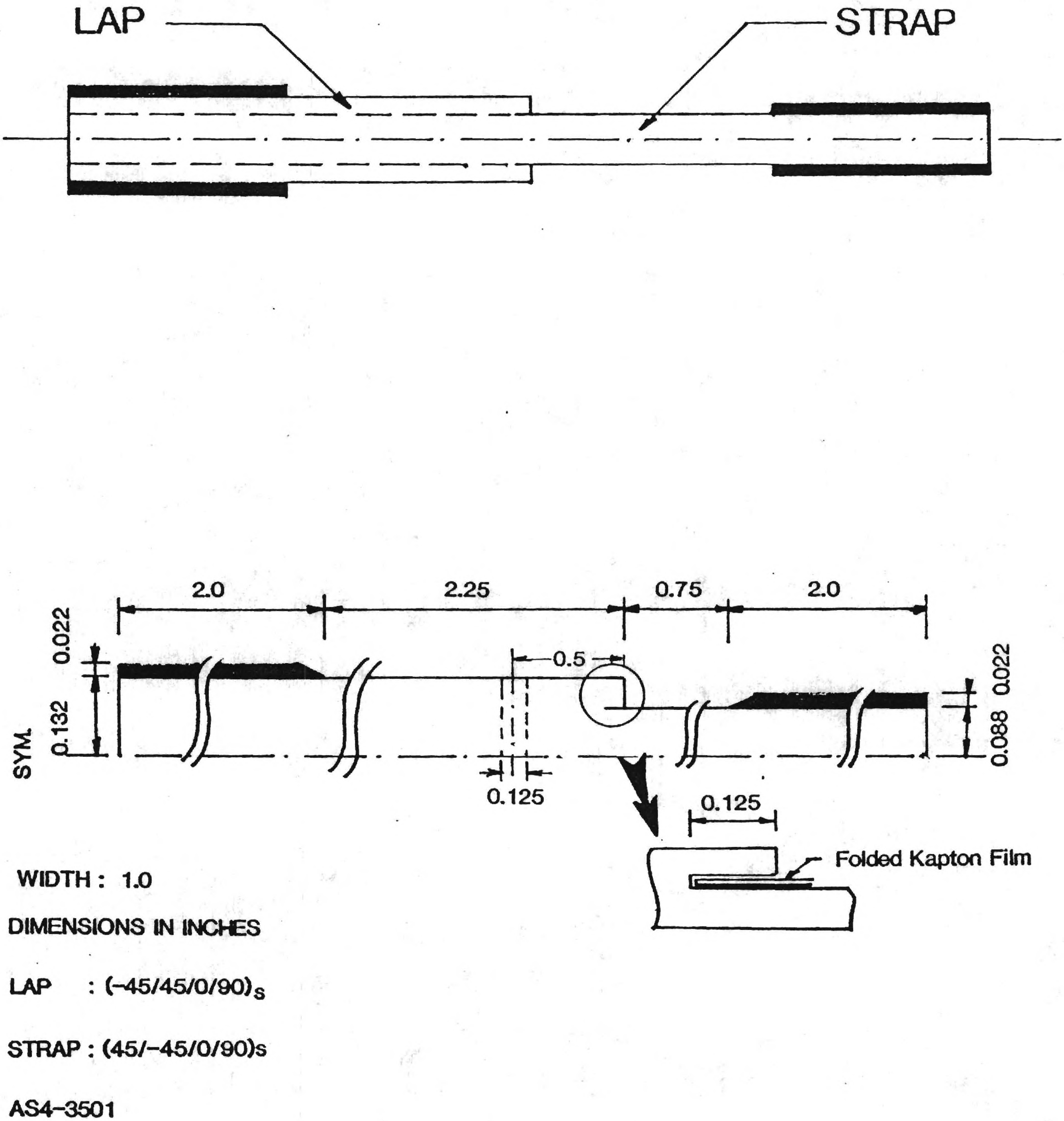
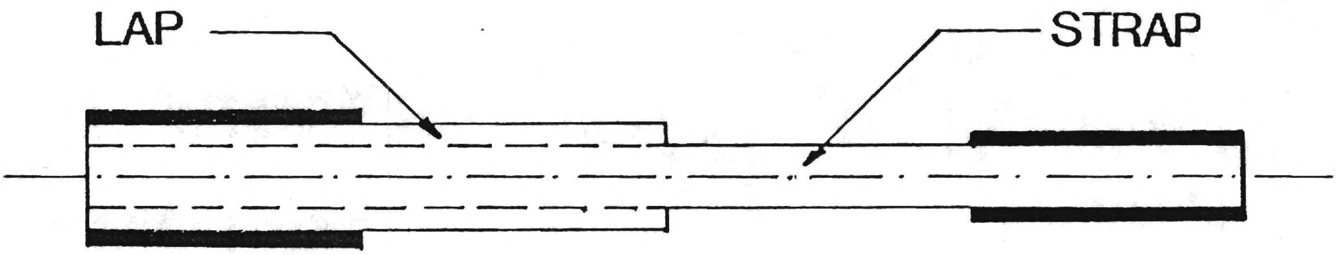
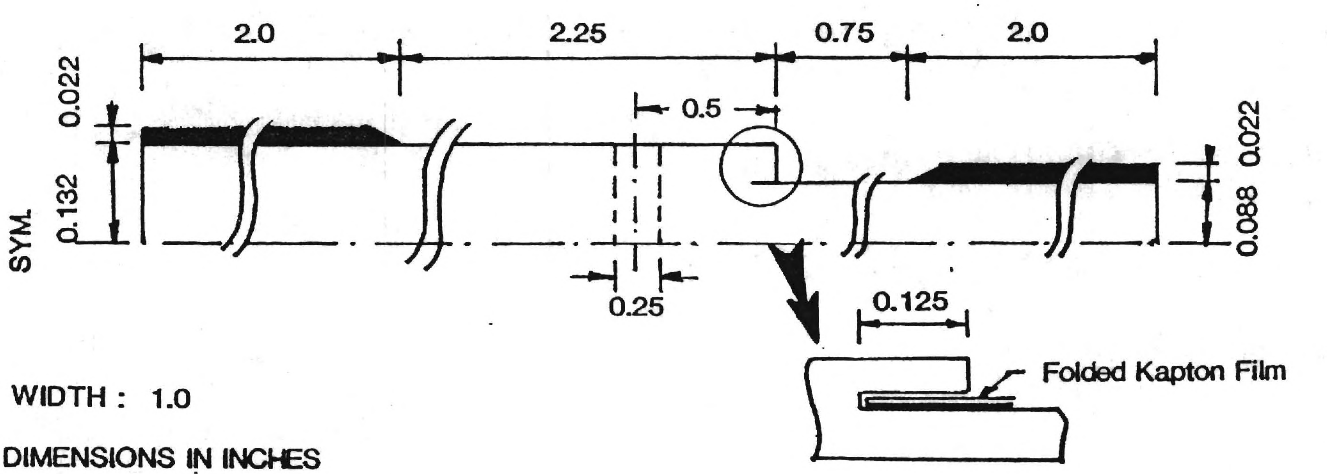


Figure 7

The Double Cracked-lap-shear Specimen
With 1/8 IN. Through Thickness Hole





WIDTH : 1.0

DIMENSIONS IN INCHES

LAP : $(-45/45/0/90)_s$

STRAP : (45/-45/0/90)s

AS4-3501

Figure 8

The Double Cracked-lap-shear Specimen
With 1/4 IN. Through Thickness Hole

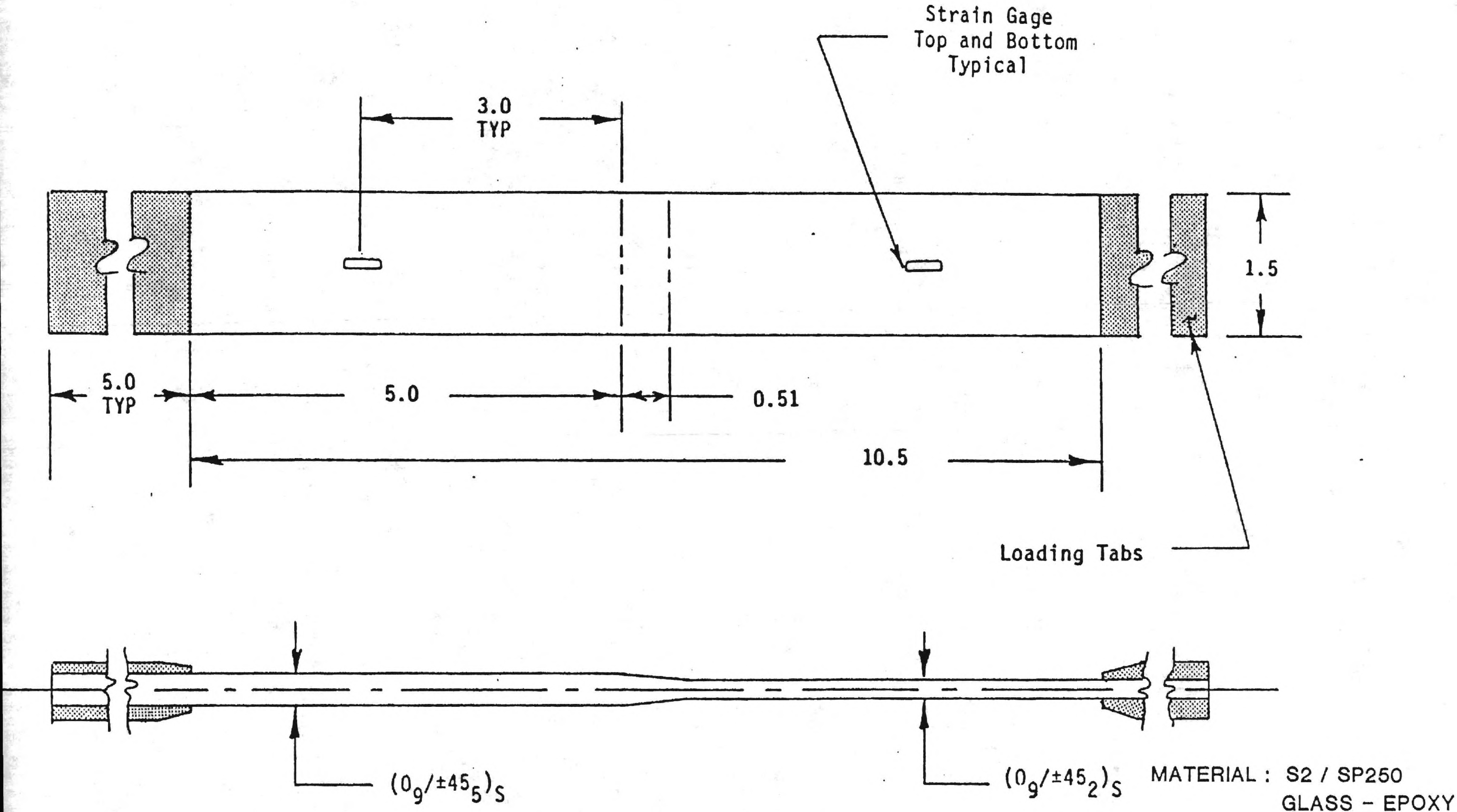


Figure 9

Ply Drop / Taper Specimen

Figure 10

COMPLIANCE FAILURE COMPARISON

SPECIMEN	DESCRIPTION	COMPLIANCE TEST $(IN/LB)X10^6$	COMPLIANCE DATA ANALYSIS $(IN/LB)X10^6$	% DIFFERENCE
0	NO DAMAGE	0.920	0.918	0.2
1	NO DAMAGE	0.681	0.644	5.5
2	LATERAL HOLES	2.424	2.385	1.6
3	LATERAL HOLES	2.597	2.569	1.1
4	THROUGH THICKNESS HOLE	1.925	1.863	3.2
5	THROUGH THICKNESS HOLE	2.000	1.987	0.7
6	TAPER	4.889	4.576	6.4

In a design environment, the designer may establish an allowable strain level. The (secant) compliance at failure and the allowable strain permits a failure load to be predicted.

CONCLUDING REMARKS

In response to the pressing need for a damage tolerance analysis methodology for composite structures, a new framework of a phenomenological type is presented and supported by data from seven experiments. The approach appears promising and further investigation is justified.

APPENDIX II

UNDERSTANDING AND PREDICTING SUBLAMINATE DAMAGE MECHANISMS IN COMPOSITE STRUCTURES

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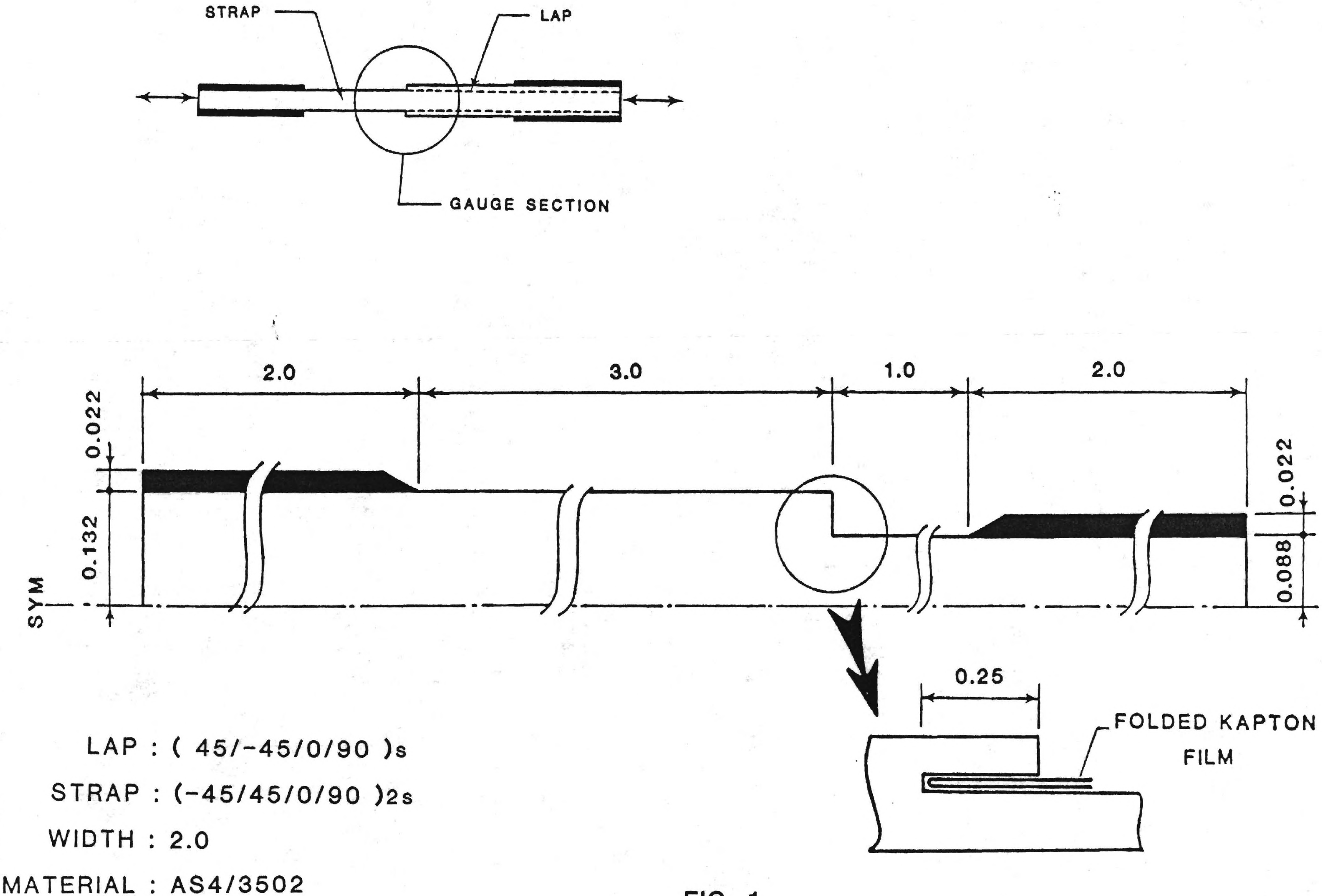
INTRODUCTION

This work originated from two sources. The first was the desire to create a damage tolerance analysis capability for composite structures --- one that is comparable to its fracture mechanics based counterpart for metallic structures. The second was the paradox posed by some delamination experiments that we performed several years ago. In responding to these inquiries a new conceptual framework for the analysis of damaged laminated composite structures has been created. The method isolates the parameters controlling the initiation and propagation of the damage modes. These include delamination, matrix microcracking and fiber fracture as well as their interaction. Moreover, a rational physical explanation of the observed behavior and quantitative agreement with test results are achieved.

SUMMARY OF EXPERIMENTAL FINDINGS AND EARLY INVESTIGATION

The present research in sublaminate damage mechanisms had its origin in the development of a new design analysis and testing methodology for interlaminar fracture which appear in References 1-3. A double cracked-lap-shear specimen made of AS4/3502 and AS4/3501-6 graphite/epoxy materials was designed and is shown in Fig. 1. The specimen layup is quasi-isotropic, balanced and symmetric with $[\pm 45/0/90]_s$ in the lap portion and



STRAP
LAP
GAUGE SECTION
2.0
3.0
1.0
2.0
0.022
0.132
SYM
0.022
0.088
0.25
FOLDED KAPTON FILM
LAP : (45/-45/0/90)s
STRAP : (-45/45/0/90)2s
WIDTH : 2.0
MATERIAL : AS4/3502

FIG. 1

$[\pm45/0/90]_{4s}$ in the strap portion. The lap interface is at ±45° orientation to the loading direction. A fundamental feature of the designed specimen is its ability to be tested under net tensile and compressive loadings. The specimen exhibits mixed-mode or Mode II behavior depending on the loading direction. As a result of the practical configuration selected for the specimen, the test results revealed some new and intriguing phenomena. Under tension loading, delamination is characterized by three stages: an initiation at lower values of the applied load followed by a stable phase where crack growth is only possible under increasing load and a final unstable terminal fracture at higher load values. Under compressive loading no crack growth is observed prior to a single, unstable, catastrophic fracture event which fails the specimen. Also, loads corresponding to failure by unstable fracture under tension and compression testing are nearly the same. The average value of the total energy release rate at initiation is 2.1 in-lb/in^2 and 4.2 in-lb/in^2 at final failure under tensile loading. Under compression the average value corresponding to initiation is 4.6 in-lb/in^2.

The increasing resistance to crack growth under tensile loading was quite puzzling since the matrix materials are brittle and tests of unidirectional single cracked-lap-shear specimen reported by Russell[4] and Wilkins[5] did not show resistance behavior. The single cracked-lap-shear specimen of Reference 4 is made of AS1/3501-6 graphite/epoxy material. The layup is unidirectional with three plies in the lap and strap portions. In Reference 5, the specimens are unidirectional with 4 plies in the lap and 10 plies in the strap. The material is T300/5208 graphite/epoxy. Our own tests performed on $[\pm45/0/90]_s$ quasi-istropic single cracked-lap-shear specimens with 8 plies in the lap and 40 plies in the strap regions show resistance behavior also. These tests confirm that the difference in behavior is not associated with the type of specimen used but rather on the layup.

In order to explain the resistance phenomenon, a systematic approach was utilized. The factors reported earlier to influence delamination behavior such as fiber bridging, fiber breakage, hygrothermal effects and curing stresses were examined to assess their effect in the double cracked-lap shear specimen.

Fiber bridging occurs between plies of similar orientation for specimens exhibiting Mode I behavior. Interfaces separating plies of the same orientation are prone to fiber nesting during fabrication. In this case, delamination resistance is increased as a result of fibers "bridging" between plies. The ±45/-45 interface at the delamination front in the double cracked-lap-shear specimen prevents fiber nesting. Also, the designed double cracked-lap-shear specimen exhibits a mixed-mode behavior with 68 percent Mode II, and no bridging was observed.

Moisture and curing stress effects can be significant in a laminated composite as the coefficients of thermal and moisture expansions are orientation dependent. However, the balanced and symmetric quasi-isotropic layup used in the designed double cracked-lap-shear specimen tends to minimize these effects.

A comparison of the photomicrographs of the fracture surfaces in the stable and unstable regions of the specimen show little or no fiber breakage. However, the fracture surface in the stable growth region is characterized by matrix microcracks. Their presence modifies the local stiffness at the crack front. How important are the effects of these microcracks? Can this effect raise the initiation total energy release rate from 2 to 4 in-lb/in^2?

A preliminary answer to these questions can be found from an investigation done on the effects of Mode I suppression[3]. Test results on double cracked-lap-shear specimens where the opening mode is suppressed through a clamping fixture show an increase in the initiation from 2 to 4 in-lb/in^2. A striking result is the fact that the clamping force needed to suppress Mode I is one percent of the applied tensile force. This may indicate that although matrix microcracking has a small effect on the overall stiffness of the specimen it may modify the local stiffness resulting in a reduction of the opening mode at the delamination front. This localized stiffness modification can be as effective as the one-percent clamping force.

TOWARD A GENERALIZED ANALYSIS OF DAMAGE MODES

The decision to select a practical configuration in order to investigate the delamination problem provided the opportunity to learn that other damage modes can develop simultaneously. Their interaction can be beneficial if the damage mechanism is understood and controlled.

The damage modes at play here can be expressed in terms of the following:

(1) Matrix microcracking controlled by the resin controlled tranverse strain to failure (ε_c);

(2) Delamination controlled by the fracture toughness (G_c), and

(3) Fiber breakage controlled by the fiber strength.

The first two modes interact to produce the resistance behavior observed in the double cracked-lap-shear specimen under tension. As matrix microcracking occurs, the opening and sliding deformations of the delamination front and the matrix stiffness are altered at the local scale. At the global scale, the total energy release and the laminate balance change. The modeling of these phenomena requires the interaction between micro and macro scales. Final failure results when delamination reaches the total length of the lap or when complete fiber breakage occurs.

The quantitative assessment of resistance is based on an engineering intuitive approach. Matrix microcracking is induced by the strain gradient at the crack front. A prerequisite for determining the strain distribution at the crack front is a higher-order theory that includes shear deformation and transverse strain as well. Delamination onset is determined using a fracture mechanics approach based on the total energy release rate and the energy release rate components associated with Mode I (G_I) and Mode II (G_{II}). Mode III (G_{III}) component is negligible in this type of specimen. The interaction between matrix microcracking and delamination is determined from the loads required to initiate each damage mode separately. This is illustrated in the flow chart of Fig. 2.

At a given value of the applied load, the strain at the delamination front and G_I, G_{II} and G_T are determined from a finite element analysis using EAL (Engineering Analysis Language) code. The element used is the four node constant strain quadrilateral element. A schematic representation of the finite element mesh is shown in Fig. 3. The number of degrees of freedom used is 5490. In order to assess the accuracy of predictions at the delamination front, the axial stress distribution is plotted on a logarithmic scale as shown in Fig. 4. This is a direct method to extract the order of the singularity in the near-field stress distribution. At this stress level the load required to initiate matrix microcracking (P_M) is determined.

The load required for the onset of delamination (P_D) is determined from G_I, G_{II} and G_T using the following law

$$G_{oc} = \xi^n G_{Ic} + (1-\xi)^n G_{IIc} \tag{1}$$

where G_{oc} is the strain energy required for the onset of delamination, G_{Ic} and G_{IIc} are the fracture toughness associated with Mode I and Mode II, respectively. These are approximately 1 in lb/in^2 and 4 in lb/in^2 for AS4/3501-6 graphite/epoxy material. The ratio (G_I/G_T) is denoted by ξ. The exponent n is material dependent[4]. Limited correlation with experimental data indicates that a value of 2 gives satisfactory results for the material used. As a check, the onset of delamination for the designed

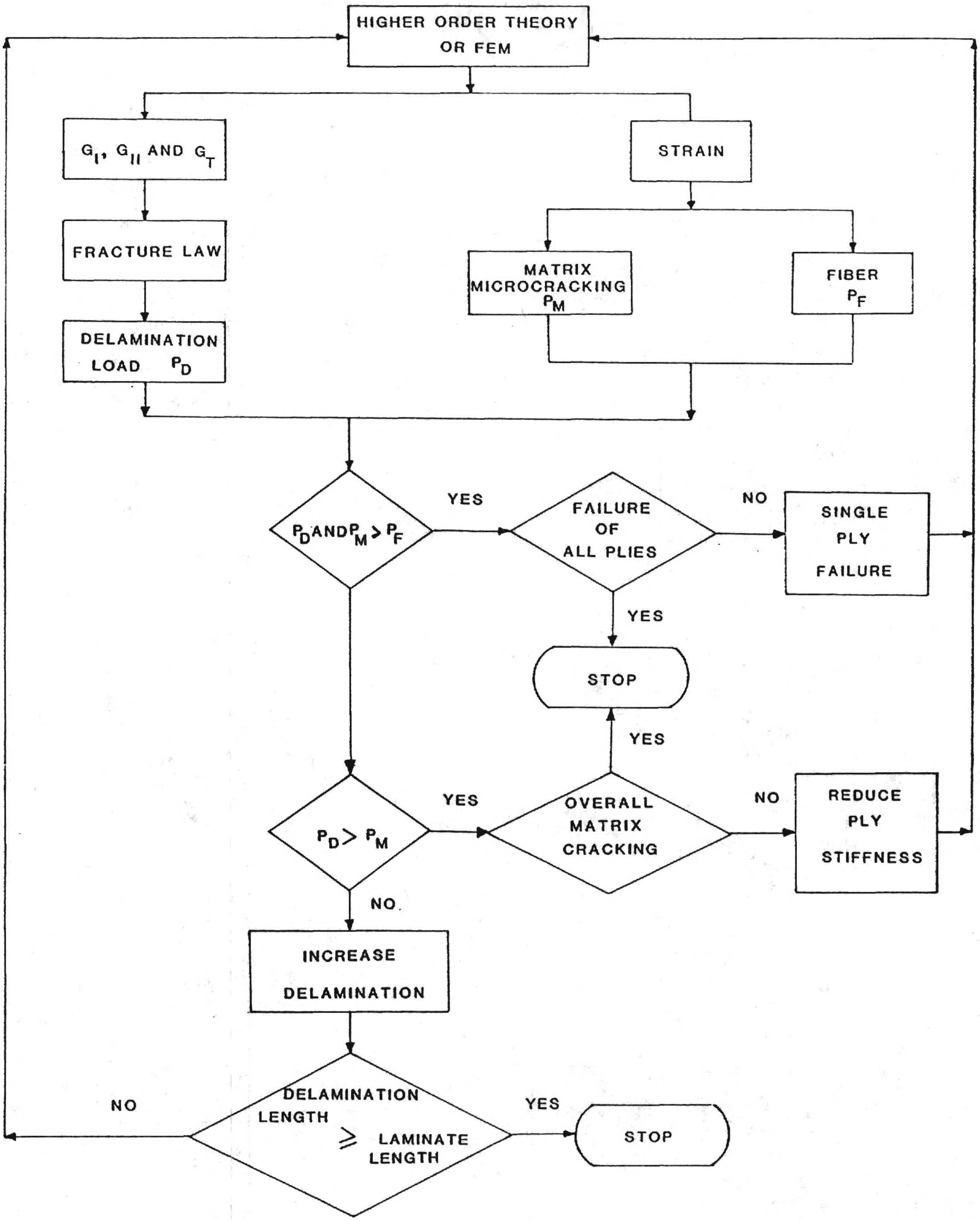
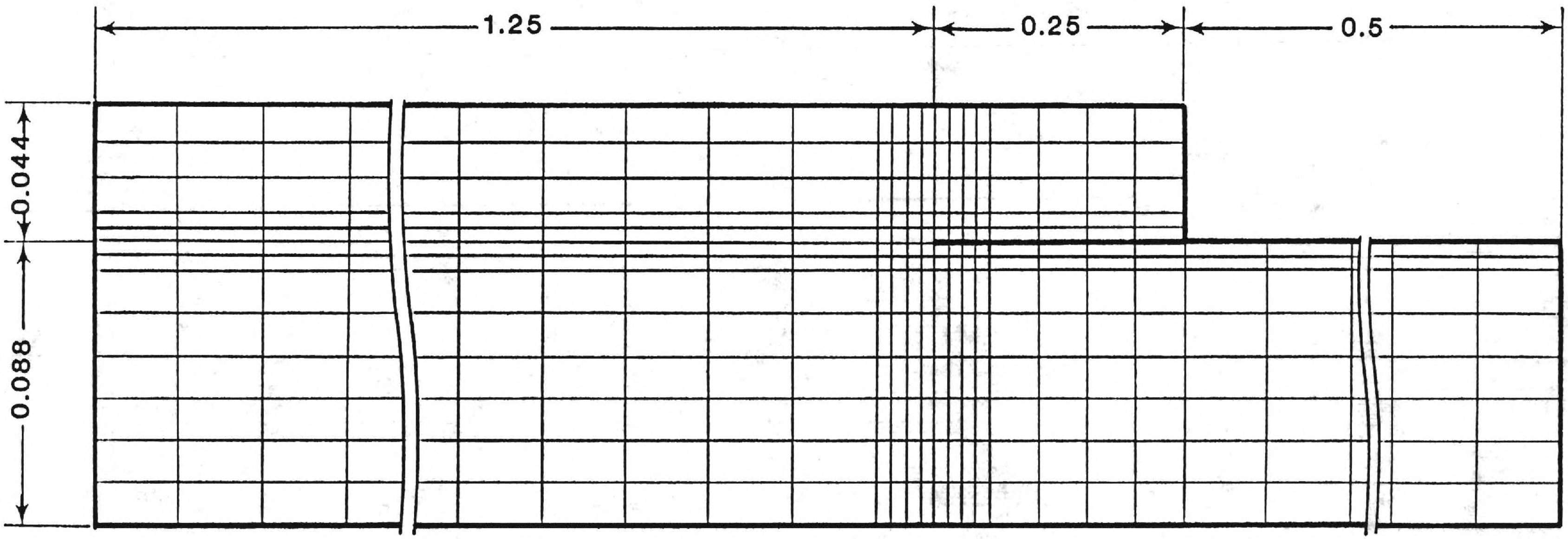

HIGHER ORDER THEORY OR FEM
G_I, G_{II} AND G_T
STRAIN
FRACTURE LAW
MATRIX MICROCRACKING P_M
FIBER P_F
DELAMINATION LOAD P_D
P_D AND $P_M > P_F$
YES
FAILURE OF ALL PLIES
NO
SINGLE PLY FAILURE
YES
STOP
YES
$P_D > P_M$
YES
OVERALL MATRIX CRACKING
NO
REDUCE PLY STIFFNESS
NO
INCREASE DELAMINATION
NO
DELAMINATION LENGTH $\geq$ LAMINATE LENGTH
YES
STOP

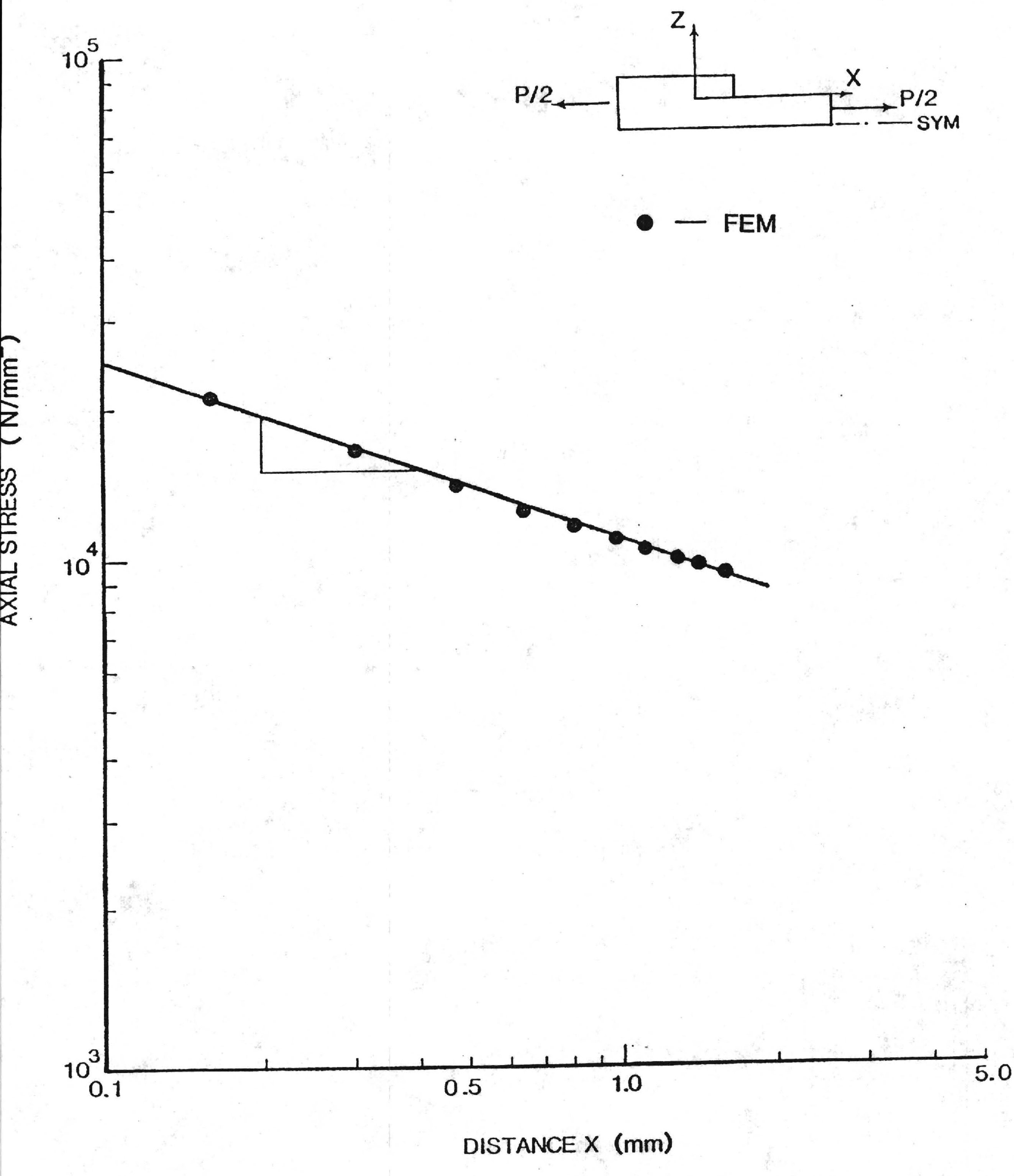
FIG. 2



NDOF : 5490

DIMENSIONS IN INCHES

FIG. 3



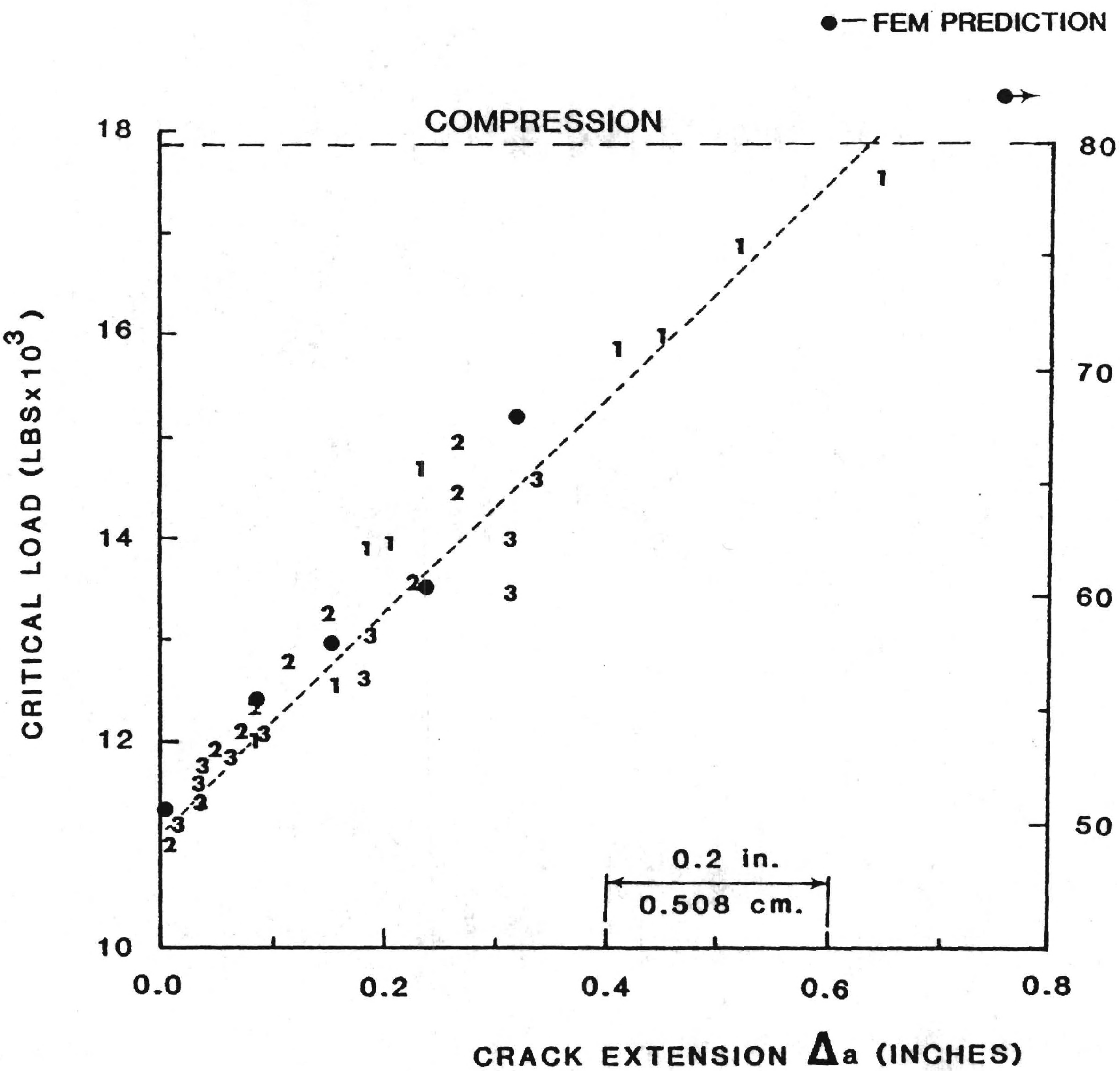
Z
X
P/2
P/2
SYM
● — FEM
10^5
10^4
10^3
0.1
0.5
1.0
5.0
AXIAL STRESS (N/mm²)
DISTANCE X (mm)

FIG. 4

double cracked-lap-shear specimen (with $G_{II}/G_T = 0.68$), is estimated at 1.95 in lb/in^2 using equation (1).

The driving damage mode is determined by comparing P_D, P_M and the load corresponding to fiber breakage (P_F). For the designed double cracked-lap-shear specimen under tension loading the driving damage modes are delamination and matrix microcracking. Under compressive loading, delamination is the prevalent damage mode. This situation is depicted in Fig. 5. The load corresponding to a crack extension Δa is denoted by critical load in figure. The solid dots are the analytical predictions. Numbers 1, 2 and 3 correspond to data points from three double cracked-lap-shear specimens generated from the same parent panel.

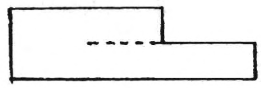
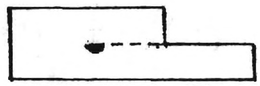
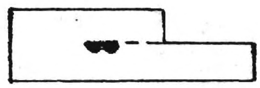
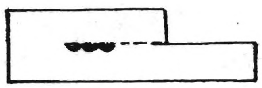
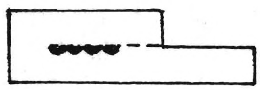
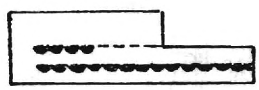
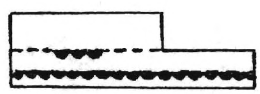
The numerical values represented by the solid dots in Fig. 4 appear in tabulated form in Fig. 6. A schematic of the prevalent damage modes appear in figure. Delamination is represented by a dashed line while matrix microcracking corresponds to solid semi-circles. Matrix microcracking occurs in the -45° ply in the strap portion at the delamination interface. The 90° plies close to the delamination front exhibit microcracking in the final stages of the resistance curve. A photomicrograph from the lap portion of a failed specimen showing evidence of matrix microcracking appears in Fig. 7. Matrix microcracking occurs at a direction normal to the fiber orientation in the -45° ply. Enhanced x-rays of two failed double cracked-lap-shear specimens tested under tensile and compressive loadings, respectively, indicate the presence of matrix microcracking in the tension specimen while no matrix microcracking appears in the specimen tested under compression.



● — FEM PREDICTION
COMPRESSION
CRITICAL LOAD (LBS x 10^3)
(NEWTONS x 10^3)
CRACK EXTENSION Δa (INCHES)
0.2 in.
0.508 cm.
18
16
14
12
10
80
70
60
50
0.0
0.2
0.4
0.6
0.8

FIG. 5

PREDICTIONS OF DAMAGE GROWTH

LOAD (LBS.)	CRACK LENGTH (INCHES)	% G_{II}/G_T	DAMAGE
11,486	0.44	60	
12,528	0.50	60	
13,020	0.61	60	
13,530	0.67	60	
15,300	0.76	66	
18,301	1.20	85	
18,301	-		

MATRIX CRACKING

--- DELAMINATION

FIG. 6

Under both loading conditions the analysis predicts no fiber breakage. This correlates with the photomicrographs of the failed specimen showing little or no fiber breakage[1,2].

CLOSING REMARKS

The decision to select a practical laminate has proved to be necessary to understand toughening mechanisms in real structures. Delamination actually does not occur in isolation---it is accompanied by other damage modes. In the present context the interaction between the damage modes is beneficial as it results in a resistance to delamination growth raising the energy onset value from 2 to 4 in-lb/in^2.

This finding establishes a new basis for the design of laminated composites---toughening can be designed-in using the interaction of the damage modes that exist in practical structures. The outcome is an efficient damage tolerant structural concept where the micro and macro scale characteristics of a composite are fully utilized. This concept can be extended to other material systems such as metal matrix composites. The prerequisite is a fundamental understanding of the damage modes at play. The framework can be based on the analysis methodology outlined in this work. The final outcome is a control of the damage modes and the use of the composite structure to its full potential.

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